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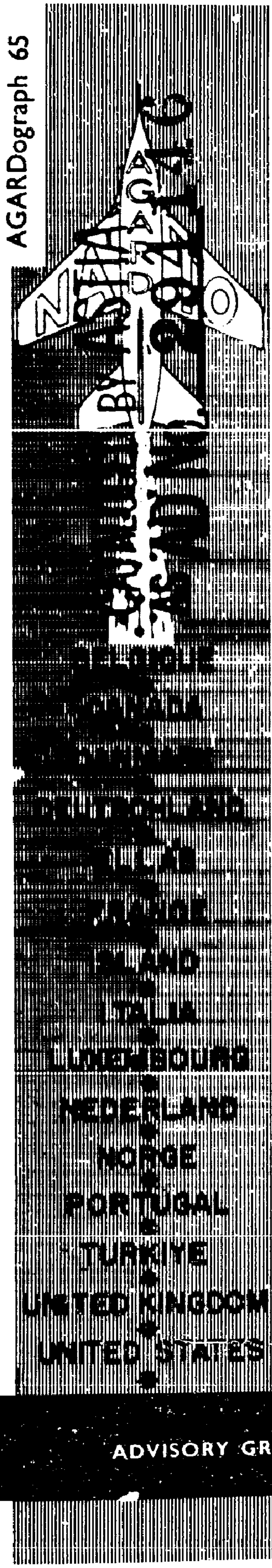

AGARDograph

STRUCTURAL ASPECTS OF ACOUSTIC LOADS

by

B. L. CLARKSON

SEPTEMBER 1960

NORTH ATLANTIC TREATY ORGANIZATION
ADVISORY GROUP FOR AERONAUTICAL RESEARCH AND DEVELOPMENT
64 rue de Varenne, Paris VII

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This Study was sponsored by the AGARD Structures
and Materials Panel

FOREWORD

This AGARDograph presents the results of a research co-ordination programme, on the structural aspects of acoustic loads, initiated by the Structures and Materials Panel of A.G.A.R.D. It had become apparent to the Panel that much work on the various aspects of acoustic fatigue had been done in many aircraft companies and research establishments but little had been done to bring the various aspects together and assess progress towards an overall solution. This work aims at presenting a summary of the present information on the various topics relevant to acoustic fatigue and also discusses the state reached in the development of the subject. It is based on visits carried out during the latter half of 1959 and the first half of 1960. The first draft report submitted September 1960 has been modified somewhat as a result of discussions in the Panel and comments by people working in the field.

As a result of the review various areas needing further study have been brought out.

B.L. Clarkson

SUMMARY

A review of the experimental and theoretical work on the various structural aspects of acoustic loads is given. The study of the basic mechanism of fatigue itself is outside the terms of reference of this review but test procedures in acoustic environments are considered. In the majority of aspects it is still necessary to rely heavily on experimental data and in general the scaling of results has not been very reliable.

The sound pressure levels in the near field of jets and rockets have been investigated experimentally for a range of operating conditions on a few engines. Extrapolation of these data to other engines shows a rapid decrease in accuracy as the difference in the two engine conditions increases. More accurate extrapolation procedures will depend on theoretical advances.

The considerable amount of data on boundary-layer pressure fluctuations establishes that the skin root-mean-square pressure level up to transonic speeds is approximately equal to 0.006 q and that the spectrum is probably sufficiently well defined for structural response estimates. There is some doubt about the r.m.s. levels at supersonic speeds and further tests are now in progress to resolve this point.

A large section of the work reported is devoted to the evaluation of structural response to random pressure fluctuations. Theories have been developed for the response of simple structures to noise but none is directly applicable to practical structures because of the lack of normal mode and other data. Some limited investigations on the types of modes being excited have begun but a much more extensive programme of tests on different types of structure is necessary. The effective use of additional damping compounds and the assessment of basic damping in a structure depend on a knowledge of the modes being excited. A variety of test procedures and design philosophies have been adopted. As it is difficult to say in the present state of knowledge which is the most effective, further work to study these procedures is required.

It has been impossible to present a thorough review of work on the effect of noise on internally stowed equipment and it is suggested that a further study be made of this particular field.

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SOMMAIRE

L'auteur passe en revue les travaux expérimentaux et théoriques effectués sur les divers aspects structurels des charges acoustiques. L'étude du mécanisme fondamental de la fatigue elle-même n'entre pas dans le cadre de cette revue, mais les procédures d'essai dans des ambiances acoustiques sont examinées. Pour la plupart des aspects il est toujours nécessaire de tenir compte de façon sensible des résultats expérimentaux obtenus et la réduction de ces résultats n'a pas été en général très satisfaisante.

Les niveaux de pression sonores dans le champ proche des moteurs à réaction et des fusées ont fait l'objet de mesures expérimentales pour des régimes divers. L'extrapolation de ces données à d'autres moteurs révèle une diminution brusque de la précision suivant augmentation de la différence entre les régimes des deux moteurs. Des procédés d'extrapolation plus précis seront fonction des progrès théoriques réalisés.

Les nombreuses données dont on dispose sur des fluctuations de pression de la couche limite permettent d'établir que la valeur efficace du niveau de pression du revêtement jusqu'aux vitesses transsoniques est approximativement égale à 0,006 q et que le spectre est probablement suffisamment bien défini pour permettre des évaluations de réponses de structure. Il existe quelque doute quant à la valeur efficace des niveaux de pression aux vitesses supersoniques et une nouvelle série d'essais sont actuellement en cours pour résoudre ce point.

Une grande partie des travaux présentés ici se consacre à l'évaluation des réponses structurelles à des fluctuations de pression aléatoires. Des diverses théories ont été élaborées en ce qui concerne la réponse au bruit des structures simples, mais aucune n'en est directement applicable aux structures pratiques, par suite du manque de données sur les modes propres et d'autres données. Quelques études restreintes sur les types de modes excités ont été lancées; toutefois, un programme de tests beaucoup plus étendu à effectuer sur des structures de type divers est nécessaire. L'emploi efficace de compounds d'amortissement supplémentaires et l'évaluation de l'amortissement fondamental dans une structure sont liées à des connaissances sur les modes excités. Une multiplicité de procédures d'essai et de philosophies de conception a été retenue. Etant donné la difficulté, dans l'état actuel des connaissances de préciser laquelle de ces procédures s'avère la plus utile, il faudra prévoir de nouveaux travaux pour l'étude de celles-ci.

Puisqu'il n'a pas été possible de faire le point de tous les travaux réalisés sur l'influence du bruit sur les matériels de bord, il est suggéré d'entreprendre une nouvelle étude de ce domaine particulier.

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B.L. Clarkson*

SECTION I

NOISE LEVELS AND SPECTRA OF JETS AND ROCKETS

1.1 INTRODUCTION

The theoretical study of the noise produced by jets and rockets based on Lighthill's original work[1,2] has not reached the stage where the noise levels can be calculated for a given engine. Several theoretical lines of approach are being persued but none looks like being able to yield results of a quantitative nature for the near field in the immediate future. It is possible, however, that more realistic scaling laws may result. The general trends of the variation of noise levels around the efflux and the significant parameters are understood. Much of the theoretical work considers the simpler case of the noise in the far field, not the more complex but structurally important near field. Consequently, data for structural response estimates must be obtained from direct measurements on the engine to be used or by extrapolation from measurements on a similar engine.

Most of the work reviewed in this section is of an experimental nature aimed at the accumulation of data on the characteristics of the near field of specific engines. An attempt is made to produce generalized intensity patterns and frequency spectra by comparing the various full-scale results. The agreement, however, whilst fair in qualitative terms, is poor by structural load standards. For an overall accuracy better than about 3 db it is still necessary to have measurements on the actual engine to be used or on one running at very similar operating conditions.

It is convenient to consider jets and rockets together in this one section as the mechanism of noise production is very similar in both cases. Recent measurements suggest that most of the noise produced by a rocket emanates from a region some 20 diameters downstream of the nozzle, i.e. where the flow becomes subsonic. Thus the main difference between the jet and the rocket would appear to be in the position of the main noise sources. This rough picture is modified by the noise which is produced in the supersonic region by the interaction of shock waves with eddies and hot spots.

A second order of complexity is introduced by such practical conditions as reverse thrust, multi-jet configurations and ground and structural reflections. Here it has only been possible to quote a small amount of empirical information.

1.2 EXPERIMENTAL AND THEORETICAL WORK

The majority of the measurements of near-field noise have been carried out by the engine companies and research establishments in free-field conditions. The aircraft

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companies have then generally used these free-field measurements to estimate pressure loads on the structure. Boeing (and Sud-Aviation) appear to be the only companies to have made comprehensive measurements on their particular configurations. Theoretical work is being carried out mainly in the Universities and post-graduate Institutes.

1.2.1 R.A.E.

The major part of the programme of noise measurements on a Ghost and an Avon engine with reheat has been completed and reported to A.G.A.R.D[3]. In addition to this comprehensive set of near-field noise levels and spectra some correlation measurements have been made by Franklin[4]. These latter measurements of the correlation of overall noise pressures have not been analysed in filter bands. Thus they give a picture of relative correlation lengths but do not give the filter band information required for structural response estimations.

An extensive series of measurements of noise levels and spectra was made on a 1/60th-scale (cold) and 1/16th-scale (hot) model of the Blue Streak rocket in its launching duct. Measurements were made with the model fully in the duct, half out, and in a free-field. In the unlined duct the reverberant conditions gave rise to an increase in noise pressure on the body of approximately 20 db. With suitable absorbent lining this increase was only 10 db.

1.2.2 Rolls-Royce, Hucknall

A comprehensive series of measurements of noise levels and spectra has been made in the near field of an Avon R.A.26 engine with a standard conical nozzle of 20.8 inches diameter[5]. Jet velocities of 1300, 1575, 1800 and 1985 ft/sec were used to enable the variation of level with jet velocity to be determined. Free-field measurements were made at radial distances in the range 5 ft - 50 ft from the nozzle and at angular positions in the range 15° to 120° from the jet centre line. Spectra were measured in 1/3 octave bands in the frequency range 35 c.p.s. to 9000 c.p.s.

In the structurally important frequency range 150 c.p.s. - 600 c.p.s. more detailed studies were made; these can be summarised as follows:-

(1) Variation in level with angular position (fixed radius and velocity)

(2) Variation in level with radial distance (fixed angle and velocity)

(3) Variation in level with jet velocity (fixed angle and radius).

These results are shown graphically in Figures 1(a), (b) and (c).

The scaling of these results to apply to other engines with different nozzle area and jet density has not been investigated experimentally. On the basis of Lighthill's dimensional analysis[1,2] it is suggested that for a given position in the near field the intensity be taken as being proportional to

$$\frac{\rho_1^2 A V^n}{\rho_0 c^5}$$

where ρ_0 is the density of the surround medium
c is the speed of sound in the surrounding medium
ρ_1 is the jet density
A is the nozzle area
V is the fully expanded jet velocity
n is the velocity index as determined from the measurements (equal to 8 in the far field).

An I.B.M. computer programme has been developed to convert these free-field measurements into pressures on a particular aircraft configuration on the assumption that the noise field is axi-symmetric, and a factor of 1.5 is used to allow for reflections. Using this method reasonable agreement was obtained between estimated and measured pressures on the Caravelle rear fuselage in the important structural frequency range of 150 - 600 c.p.s. At higher frequencies the agreement was apparently not quite so good.

A smaller scale test rig is also available but is used mainly in the development of suppressors for far-field noise reduction. The engine is a Blackburn Palas 600 having a nozzle diameter of 5.52 inches and a velocity range up to 1670 ft/sec. Near field measurements were made on the corrugated nozzle for the Avon engine installation of the Comet.

A model air jet installation is in the course of construction using a heated air supply. This rig will be used for the assessment of suppressor configurations at high pressure ratios.

With the increasing use of by-pass or ducted-fan types of engines in rear-mounting configurations the possibility of structural damage forward of the intake may have to be considered. Rolls-Royce have made measurements close to the intakes of several engines but much of this work is only available in company confidential reports. The general position seems to be that there is broad-band noise on to which are superimposed one or more very pronounced discrete frequencies due to the compressor. These discrete frequency components are generally in the high frequency range (above about 2000 c.p.s.) and are therefore more likely to be significant from the point of view of passenger comfort rather than structural fatigue. Measurements indicate that the broad-band noise level close to the intake would be of the order of 130 db if there were no discrete frequencies present.

1.2.3 Southampton University (Department of Aeronautics and Astronautics)

A theoretical and experimental programme is aimed at the study of noise production by a turbulent mixing region[6]. Turbulence and noise measurements are being made on subsonic cold air jets from 1 in. to 4 in. diameter.

Recent theoretical work[7] analyses a convected field of turbulence with respect to moving axes and relates the spectra in the moving reference frame to the noise output. The basic theory of noise production by a turbulent mixing region as propounded by Lighthill requires a knowledge of these moving frame spectra. Hot-wire measurements made to date at N.A.C.A., Cleveland and elsewhere have been for fixed-measurement positions. To obtain the noise-producing component of the velocity fluctuations

seen by a moving observer in the turbulent region it is necessary to use two hot wires, one downstream of the other. A time delay is inserted in the signal from the upstream wire equal to the time of convection of the stream from the first wire to the second. Comparison of the delayed signal with the signal from the second wire now indicates the change which has taken place in the original signal in the moving frame of reference. Hot-wire measurements of turbulence in the mixing region of a subsonic model jet have been made in this manner; and the spectra of the velocity fluctuations on the moving frame of reference have been obtained by cross-correlation techniques. The limitations of this method in a region of changing velocity (shear) are being studied.

The theoretical analysis to date is restricted to homogeneous turbulence convected with a uniform velocity but some of the results are applicable to shear flows. Further refinement of the theory is in progress.

With a view to understanding more fully the mechanism of noise reduction, a parallel investigation of the turbulence and noise field of a circular nozzle containing a single triangular wedge has recently been initiated.

Due to the increasing importance of compressor noise, particularly from the subjective point of view, an experimental project on the study of noise from a small-scale axial compressor has been started although no results are available yet.

A series of near-field noise pressure correlations is being made on a 2 in. cold air jet with a view to understanding the correlation patterns produced by a jet[8]. Cross-correlation measurements between two probe microphones are being used to trace back to the source area of particular frequency bands in the mixing region. It is hoped to be able to build up in this way a mathematical model of the distribution of the sound sources and their position in the jet stream and to relate this to the measured correlation pattern in the near field.

Full-scale pressure correlation measurements have been made on a tailplane test specimen at the Royal Aircraft Establishment, Farnborough. The specimen is positioned 3.5 diameters downstream and 1.4 diameters out from the jet boundary of a de Havilland Ghost engine. The narrow band pressure correlations on the tailplane surface in the longitudinal (parallel to the jet axis) and lateral directions are shown in Figures 2(a) and 2(b). From these results it can be seen that the scale of the lateral correlations is considerably greater than the scale of the longitudinal ones. This is in line with overall correlations made by Franklin on a 2 in. cold jet[9], and also by Callaghan, Howes, Coles and Mull[10]. The implication is that there is a pseudo wavefront in the lateral direction, i.e. spanwise on a tailplane and circumferential on a fuselage. Hence testing rigs which use single sources to produce plane waves running along the structure may be a reasonable approximation to the actual condition, at least over a relatively large structural area. This is discussed in more detail under the heading of test facilities.

1.2.4 College of Aeronautics

Lilley is working on theories of noise production by turbulent mixing regions[11]. He has produced a simplified theory of jet noise on the assumption of isotropic turbulence in a high-shear region. The characteristics of the isotropic turbulent field

are those given by Batchelor and the scale of magnitudes is based on Laurence's measurements on a model jet.

1.2.5 O.N.E.R.A.

An earlier series of measurements of the noise in the near field of the ATAR.101F engine was reported by Kobrynski[12] in 1957. Current work is concerned with the measurements of the noise pressures on the fuselage and tailplane of the Caravelle noise test specimen at Toulouse with one engine running (Avon R.A.28). These have been compared with earlier measurements on the fourth aeroplane with 2 engines running and also with the Rolls-Royce estimates from the I.B.M. computer.

A smaller test installation using a Marbore engine having a nozzle diameter of 22 cm has been set up to carry out panel tests[13]. The near-field noise pressures of this jet have been measured to enable the typical panels of the Caravelle to be placed in a region of similar noise intensity and spectra.

1.2.6 Göttingen (Max Plank Institute for Flow Research)

A theoretical and experimental study of the scattering of sound by turbulence is being carried out and has a bearing on the case where noise from one jet passes through the efflux of an adjacent jet before reaching the structure[14]. Only the case of a homogeneous turbulent field has been considered to date.

A turbulent field is set up in a duct of 30 inches diameter by the insertion of a grid into a laminar airflow of 25 m/sec. The degree of turbulence can be changed by changing the grid or by moving the grid further away from the test section. A pulse of high-frequency sound is passed across the test section and the reduction in intensity produced by the turbulence is measured. The pulse duration and the separation of pulses can be adjusted to eliminate the effect of reflection from the wall of the tube on the direct pulse. A rectangular gate with a time delay on the receiving line can be adjusted to pass the direct pulse and cut out the reflected signals. The direct pulse at the receiver is then compared with the transmitted signal to give amplitude and phase change.

The amplitude of the received pulse varies randomly about a mean value and the relative phase of the transmitted and received pulses also varies randomly. Measurements to date have been confined to a comparison of mean intensity levels - future experiments will study the statistical properties of the received pulse envelope and also the phase relationship. An analogue correlator is to be used for this work.

The reduction in intensity varies typically as frequency squared and therefore to obtain measurable reductions high frequencies are used. The range of frequencies in this experiment is 50,000 c.p.s. to 500,000 c.p.s., giving an additional experimental advantage of eliminating extraneous sources of noise from the tunnel fan, etc. Different types of turbulence have not been fully investigated yet but the trend of results indicates that the scattering at the low frequencies (100 - 1000 c.p.s.) of major structural response will be small.

1.2.7 N.A.S.A. (Cleveland)

Earlier work forms the most comprehensive study in the U.S.A. of noise spectra and pressure correlations in the near field of a full-scale jet engine[10]. Recent work at Cleveland has been concerned with the effect of forward velocity on the near-field noise pressures[15]. Measurements were made on a single-engined aircraft, fitted with a J-33 turbo jet, at positions 2 and 3 diameters downstream along the boundary of the jet. The flight Mach number was varied from 0.35 to 0.70 at altitudes of 10,000, 20,000 and 30,000 ft. The frequency response of the microphone, recording and measuring equipment was flat from 50 to 8000 c.p.s. but the background noise level was reported to be only 6 db below the datum level.

The indications of this experiment are that the near-field noise pressure levels are proportional to a low power of the relative velocity of the jet and surrounding air and that there is no significant change in spectra within the accuracy of the measurements. The accuracy of the measurements, however, was not sufficiently great to determine whether this value was 1 or 2. The only significant reduction measured was that of the variation with altitude as shown in Figure 3. The engine parameters are not available but the trend is for the reduction in sound pressure level to be proportional to atmospheric density raised to a power of 1 to 1.5.

The indication from these results is that there will be no reduction in pressure level along the jet boundary due to forward velocity. However, measurements of the near field of stationary jets show that the velocity index increases with distance away from the jet and therefore for structure 2 diameters or so away from the jet there will be a marked reduction with forward velocity as indicated by the Caravelle strain measurements.

1.2.8 N.A.S.A. (Langley)

A series of measurements of noise levels and spectra has been made at Langley[16] on a range of small rocket motors having thrusts from 1000 lb to 5000 lb. Five of the six motors had the same propellant configuration but nozzles of different exit diameters (3.5 - 9.07 in.) to provide a range of exit pressures, velocities and Mach numbers. The sixth motor, having a smaller diameter propellant grain and a smaller nozzle (2.88 in.), was used in the detailed surveys for fixed nozzle conditions. The engines were fixed on an outdoor stand with a jet centreline 3.5 ft above the ground. Noise measurements were made with a condenser microphone having a frequency range 5 - 12,000 c.p.s.

The spatial variation in pressure level in the 1/3 octave 448 - 566 c.p.s., shown in Figure 4, indicates the general trend of the directional pattern. The indication from these curves is that most of the noise is being produced in a region some 20 diameters downstream of the nozzle. High-speed photographs of the flow indicate that this position is the point where the flow becomes subsonic.

The variation in overall level along a line parallel to the jet centreline at a distance equal to 3 diameters away is shown in Figure 5, where the axial distance is non-dimensionalised by dividing by the nozzle diameter. The results from the 3 different nozzles tested all lie reasonably closely to a single line. Also of interest is the plot on this curve of 4 points from motors having thrusts in the range 100,000 and 130,000 lb. The deviation of these points from the curve is only of the same order as the scatter of the small motor tests.

The power spectra of the noise in the near field is shown in Figure 6 at positions 3 diameters out, and 11.5 diameters downstream and 2.5 diameters upstream. The levels here have been adjusted to unit bandwidth. The spectra are very flat with a tendency to peak at about 1000 c.p.s.

A model jet facility with a 2 inch diameter nozzle and heater to give air at 1000°F is being used for model tests to find the effect of the reflection of the exhaust by a ground launch pad. The operational Mach number is 1.6.

Noise measurements have been made forward of the rocket nozzles on an Atlas missile in connection with Project Mercury[17]. The variation in overall level with distance forward of the nozzle is shown in Figure 7(a). Spot measurements from some smaller rockets are also shown on this figure and it can be seen that they all tend to lie on a reasonably smooth curve. The two solid lines for the different firing conditions are based on experimental results from model supersonic jets and small rockets.

The noise spectrum at the position of the Mercury capsule is shown in Figure 7(b). The overall level is approximately 145 db. Also superimposed on this curve is the spectrum inside the capsule. External spectra at other positions along the missile are stated to be similar in shape to that measured at the nose.

1.2.9 W.A.D.C. (Aeromedical Laboratory)

The work is mainly concentrated on far-field jet and rocket noise. Model tests are being made in an anechoic chamber 30ft x 22ft x 22ft to assess the effect of silencing devices on far-field noise.

Full-scale rocket noise measurements have been made[18] and spectra obtained by von Gierke have been non-dimensionalised and compared with the spectra of turbo-jets and model jets as shown in Figure 8(a). By a suitable choice of parameters Von Gierke has been able to collapse all the points on to a very smooth curve. The spectra are for the overall sound power level. A plot of the total acoustic power output is shown in Figure 8(b). It can be seen that the overall intensity of rocket noise is less than would be predicted by using the V^8 subsonic relationship.

The effect of deflecting a rocket efflux by the launching pad is indicated by results of model tests shown in Figure 9. The near-field overall levels are increased in each of the deflector configurations[19].

1.2.10 U.C.L.A.

Powell[20,21] has continued theoretical studies of the noise radiation from jets and rockets and also of jet instability and flow resonance phenomena. Experimental work on these topics has begun in the recently completed Aerosonics Laboratory. Theoretical work on the propagation of sound in high-speed flow, extending the multiple reflection theory to finite frequencies and ducts of arbitrary shape, is also in progress.

1.2.11 Bolt Beranek and Newman (B.B.N.)

Measurements have been made of noise levels and spectra in the near field of turbo-jets and large liquid propellant rockets[22]. Typical spectra and pressure correlations

on a missile surface are shown in Figures 10 and 11. The experimental pressure correlation (in octave bands) is compared with a theoretical prediction based on the assumption that the noise originates at a single point and propagates upstream along the axis of the missile. As would be expected, the experimental curves decay more rapidly than the theoretical ones because of the random nature of the source area. Indications are that the distribution of pressure amplitude is approximately Gaussian in spite of the effects of non-linearities of the air at the very high noise levels.

The work on turbo-jet noise has resulted in a prediction procedure[23] discussed in detail later. The basis of this is the extrapolation from the near-field levels and spectra measurements on an engine having a jet velocity of 1850 ft/sec and a nozzle diameter of 22 inches (N.A.C.A. Cleveland). The procedure involves the following steps:-

1. Overall level correction

$$\Delta F = 80 \log\left(\frac{v}{1850}\right) \text{ db}$$

where v = fully expanded jet velocity.

2. Angular shift of the equal-level contours determined from an empirical curve, dependent on jet velocity.

3. ΔF is added on to the reference contour pattern and the whole pattern is rotated.

4. The shape of the spectrum at any point is given by the reference spectra as a function of distance downstream of the nozzle x/D only.

A reference near-field contour pattern (Fig.12(a)) and a series of reference spectra (Fig.12(b)) are used as the starting point.

The effectiveness of this prediction procedure is discussed later (Sections 1.3.1 and 1.3.2).

A prediction procedure for the noise levels on a missile structure is also given in Reference 23. Unfortunately the author does not have sufficient rocket noise data to check the procedure for a variety of rockets and so discover the likely error.

1.2.12 Boeing

A hot model jet is used mainly for the development of suppressors. The model jet was also used to measure pressures on a model B-52 wing and when checked with full-scale results reasonably good agreement was achieved[24]. The model was then used to predict the pressures on the B-52 wing when suppressors were fitted to the engines. On the assumption that it is necessary to duplicate the jet velocity and density, size is now the only major variable to be accounted for in the scaling process. The effect of this is to give a frequency shift and a change in intensity:

$$f = f_m\left(\frac{l_m}{l}\right) \qquad \text{(jet velocity the same in both cases)}$$

$$\text{Spectrum level shift} = 10 \log\left(\frac{l}{l_m}\right)^2$$

The result of this scaling is shown in Figure 13(a). It can be seen that the spectrum peak is predicted reasonably accurately but that the model underestimates the high-frequency components of the full-scale jet. However, the structurally important bands (150 - 600 c.p.s.) are well reproduced. The corresponding wing contours derived from a model wing behind a model of one engine pair show reasonable agreement (Fig.13(b)). The effect of two pairs of jets was obtained by adding together the intensities produced by each pair separately.

The success of this programme led on to the simulation of large rocket motors. In this case it is not possible to maintain the same nozzle velocity and density between model and full scale and a scale factor on intensity, equivalent to the square root of the thrust ratio, is used. The results from this scaling are shown in Figure 14.

1.2.13 Douglas

A 1/5 scale model jet heated to 1000°F was used mainly for the development of noise suppressors. Structural development tests and full-scale noise suppressor and thrust reverser tests were carried out at the DC-8 engine test stand at Edwards Air Force Base. Scaling from model to full scale was not very satisfactory in the first three full-scale octave bands, probably because the reflecting effect of the ground was not accounted for on the model rig. The model was about 25 diameters above the ground compared to 2.5 diameters full scale. For this reason, near-field model data were not used to determine absolute values of acoustic loading.

Full-scale tests on thrust reversers showed near-field sound level contours moved forward through approximately ninety degrees. The angle of maximum noise intensity now makes an angle of about 120° with jet axis. As the reverser can be used throughout the flight regime if required this new intensity pattern may influence fatigue life.

The acoustic loads on the control surface structure of the DC-8 were estimated from near-field sound pressure level measurements made around a Pratt and Whitney Aircraft JT3C-6 engine equipped with a daisy-plus-ejector noise suppressor. Free-field contours of equal sound pressure level in octave bands were determined from these near-field measurements. The loading of a piece of structure on the aircraft was estimated from these contours by over-laying the contours on a drawing of the aircraft, one engine at a time. The contribution of four engines was determined by summing the contribution from each engine. Finally, a correction of 3 db was added to take into account the increase in sound pressure level at the surface compared to the free-field value. Calculations indicated that this correction for flat, finite panels might be as great as 9 db (depending on the size of the panel) except for the following considerations: (a) the panels are curved and not flat; (b) the sound from the distributed noise source impinges on the surface with approximately random

incidence; (c) the panels are not perfectly reflective; and (d) the wave front is not plane near the source. The 3 db correction is an engineering approximation that has been checked by experiment.

Estimates were made for the take-off condition of the Pratt and Whitney Aircraft JT4A-3 and Rolls-Royce Conway RCo-12 engines by applying a correction based on the assumption that the noise intensity varied according to AV^8 and using gas generator data supplied by the engine manufacturer. The velocity was defined as the ratio of thrust to mass flow.

These estimated acoustic loads were later checked by measurements on the control surfaces of the DC-8 equipped with JT3C-6 and RCo-12 engines[25]. In general, the estimates were within 3 db of the measurements.

Measurements have been made of the free-field noise levels around Thor and Nike Zeus rocket motors and also of the noise levels on the surface of missile structures and in the internal equipment bays. Some of these measurements are given in References 26 and 27. The noise levels inside the equipment bays were only 10 db lower than the external levels. This high internal level is presumably due to the lack of absorption in the enclosure.

1.2.14 Lockheed (Burbank)

A full-scale jet engine facility has been set up in which near-field noise spectra and correlation measurements are being made. Only provisional results were available at the time of the visit. A panel group with an overall dimension of 40 in. x 60 in. was set up in the near field and pressure correlation in filter bands were measured. Provisional correlation results looked rather inconsistent with previous measurements by N.A.C.A. Instrumentation and procedure are being checked.

1.2.15 Institute of Aerophysics, University of Toronto

Ribner is engaged on the development of a simple source theory of jet noise[28,29]. This theory is based on the observation that in low-speed turbulence Lighthill's quadrupoles combine to form simple pole sources. The source strength is

$$\frac{\partial^2 p_0}{\partial t^2}$$

where p_0 is the local pressure in the turbulence calculated as if the flow were incompressible. Directionality results in part from convection of the eddies and in part from refraction by the mean shear flow. As in the Southampton studies, emphasis is placed on the spectra as seen by an observer moving with the general flow. Considering a convected source of a single frequency, as seen by a moving observer, it has been shown that this seen by a stationary observer is a band of frequencies whose width depends on the speed/eddy size. This effect of convection broadening also occurs for convected bands of frequencies.

Directionality and spectra show sufficient qualitative agreement with experimental results to justify further refinement of the theory.

Other relevent theoretical work has been done on the strength distribution of noise sources along a jet[30].

1.3 DISCUSSION

It is seen from the above list that a considerable amount of experimental data on jet engines and rockets has been obtained. The discussion of these results will be concentrated mainly on the jet engine results because a large proportion of the rocket data is classified. The mechanism of noise production, however, is very similar in the two cases and therefore the general trends observed in jet noise should also apply to rocket noise. The lack of an adequate working theory to apply to the near-field noise of jet engines makes attempts at scaling results very innaccurate in some cases.

1.3.1 Overall Pressure Levels (Free Field)

Three comprehensive series of near-field noise measurements have been carried out on the following engines:

1.	N.A.C.A. Cleveland	9600 lb thrust 1860 ft/sec 1.85 ft diam.
2.	D.H. Ghost (R.A.E.)	5000 lb thrust 1800 ft/sec 1.75 ft diam.
3.	R.A.26 (Rolls Royce)	11000 lb thrust 1,985 ft/sec 1.73 ft diam.

As a result of the measurements on Engine 1 Franken and Kerwin have produced an empirical procedure for the estimation of near-field levels. As described in Section 1.2.11, this procedure involves a change in level proportional to V^4 plus a small rotation of the pressure contours. The rotation has the effect of slightly modifying the velocity index at any fixed point. Using this procedure based on measurements from Engine 1, estimates can be made for the near-field noise of Engines 2 and 3 and compared with the experimental results. Figures 15(a) and 15(b) show the estimates and measurements for Engine 2 at 1800 ft/sec, and 1080 ft/sec respectively. In the first case little correction is needed (only 1 db) to allow for velocity changes and the 143 db and 134 db contours are to within 1 db of the estimates. 1 db is a reasonable allowance to make for experimental errors. The predictions over-estimate the higher levels closer in to the jet by up to 5 db. At 1080 ft/sec (Fig.15(b)) the predictions involve a 19 db reduction due to velocity change and as can be seen the actual levels are 5 - 10 db higher than the estimates. This suggests that the velocity index should be lower than 4 - a fact which has been shown by Wolfe[3]. In extrapolating upwards for Engine 3 at 1985 ft/sec the prediction gives 2½ db increase on the standard contours (Fig.15(c)). Predictions for this engine are 5-10 db higher than the measurements.

The discrepancy between prediction and measurements can be partly explained by the fact that it is not possible to use a constant velocity index throughout the near

field. The theoretical value of 4 is in fact only reached at a point some 10 diameters away. Figure 16 shows the value of the velocity index obtained for Engine 2. Superimposed on this figure are the contours for Engine 3 and three spot points in the jet boundary for Engine 1. It can be seen from these curves that apart from the 3.5 contour there is some discrepancy between the indices for different engines. However, the indication that a large part of this near field area has a velocity index lower than four partially explains the difference between measurements and estimates for the previous cases. The possible error in extrapolating from one engine to another is shown[31] in Table I. The derivation of this table by McClymonds is not clear but the overall picture it gives may not be too far from reality.

Little information is available on noise levels forward of the nozzles. Although this is lower than levels aft it may be important in some designs using large engine installations. Intake noise, although generally of a higher frequency, may be important in some cases. Information here is lacking.

1.3.2 Spectra

Von Gierke has shown that spectra for the overall sound power level of a wide range of jets and rockets can be reduced to a non-dimensional form and all lie close to a single curve (Fig.8(a)). At any point in the near field the shape of the spectrum varies with position. Close to the jet the predominant factor is the distance downstream from the nozzle. Franken and Kerwin produce a method of estimating the spectrum shape in the near field on the assumption that for prediction purposes the distance downstream is the only factor affecting the shape of the spectrum. Three reference spectra are given (Fig.12(b)) for distances downstream equal to zero, ten and thirty nozzle diameters. For any point of interest in the near field the distance downstream of the nozzle is measured and the appropriate spectrum shape used. The reference shapes are those measured on Engine 1.

To check the general application of these curves they are compared with experimental data from Engines 2 and 3. Figure 17(a) shows the spectra obtained at points along a line normal to the jet axis and passing through the plane of the nozzle. The experimental points range from 2.85 to 17.2 nozzle diameters away. The points follow the general trend of the predicted curve with a tendency to be approximately 5 db down in the lower two octaves (37.5 - 150 c.p.s.). The spread of the points over the rest of the curve is of the order of 5 db with no apparent dependence on distance away from the nozzle. Figure 17(b) shows the corresponding results for the line perpendicular to the jet axis and 10 diameters downstream. In this case there is a greater spread of the points (5 - 10 db) again with a tendency to be lower than the predicted curve in the lower two octaves (37.5-75 c.p.s., 10 db down; 75-15 c.p.s., 5 db down). At the points nearest to the jet (approximately 5 diameters out from the jet centre line), the peak in the spectrum occurs at a somewhat lower frequency than predicted (about ½ octave down).

1.3.3 Overall Pressure Levels on Structures

The increase in free-field levels due to reflections at the surface of the structure has not been studied systematically. Rolls-Royce use a factor of 1.5 (3.5 db) to allow for reflections on a typical surface and claim good agreement between surface pressures estimated this way from free field measurements and actual pressures measured on the Caravelle rear fuselage. Douglas use 3 db and recent measurements on

the Caravelle indicate an increase of 4 db. The effect of reflection from the fuselage side on to the tailplane surface pressures is also another possible source of pressure increase.

1.3.4 Pressure Levels in Multi-Jet Configuration

Where the jet engines are widely separated (relative to the jet diameter), there will be little increase in the higher intensity levels due to the effect of the additional engines. But when two jets are placed close together as in the Comet and in the twin jet pods of the B-52 the effect will be to double the intensity of the noise in directions normal to the line joining the centres of the jets. However, in the other directions there will be some shielding of one jet by the other and the exact values of the increase over a single jet are not known. Work on attenuation of sound by a turbulent region is of interest in this context. Model tests have been used by Boeing with some success although the general validity of their scaling procedure must be in some doubt in the light of the difficulties in scaling the simpler case of a single jet in a free field.

1.3.5 Pressure Correlation

For structural response estimates it is necessary to know the pressure correlation in relatively narrow frequency bands. The only comprehensive series of measurements on a full scale jet are those made at the N.A.C.A., Cleveland[10]. This presents correlation measurements at several positions in the near field although most of the work refers to positions along the jet boundary. From this work it is not possible to define the trends in the near field to predict with any confidence the correlation pattern at any point of interest. Several attempts to collapse the spread of the experimental data on to a single curve with non-dimensional co-ordinates have failed. Close to the boundary of the jet the pressures are influenced considerably by the moving hydrodynamic pressures associated with the jet efflux. It is reasonable to assume that in these conditions over short distances the pressure field is essentially a convected field, and therefore the space and auto correlograms can be related through the convection velocity. Away from the jet boundary the jet forms a complex system of sources distributed over distances comparable to the distance of the point from the jet. Until the composition of the source area is understood more precisely it is not likely to be possible to predict or scale near-field correlation patterns from one engine to another.

A typical comparison which is available at the moment is shown in Figure 18. Here the N.A.C.A. Cleveland results for the longitudinal (parallel to jet axis) and lateral directions are shown for the 500 c.p.s. band at two positions along the jet boundary. It is seen that the scale of the lateral correlation curve is greater than that of the longitudinal one but the difference is greater at the upstream position. For comparison the longitudinal and lateral correlation curve at 500 c.p.s. on a tailplane structure behind a Ghost engine at two speeds is shown. The measurement position is 1.4 diameters out from the jet boundary and 3.5 diameters downstream. The general trend of the curves is the same with the lateral scale about twice as large as the longitudinal scale. However, it is clear that the jet boundary results could not be used to predict the tailplane patterns with any great accuracy.

The model experiments[8] (Sec.1.2.3) aimed at understanding the correlation patterns in the near field in terms of the statistical properties of the source region seem to offer the best chance of arriving at a scaling and prediction procedure.

Further full scale measurements on the Lockheed rig in the near-field region should provide more empirical data. These tests should also give comparisons between free-field correlations and correlation patterns on a structural surface. The indication from the N.A.C.A. results is that the correlation pattern on a plate is essentially the same as the free-field correlation. However, where reflections are significant, as in the angle between fuselage and tailplane for example, the correlation pattern may be considerably changed from the free-field case and should be investigated.

1.4 CONCLUSIONS

A study of the work reported in this section shows that considerable data on near-field pressure levels are available. From these it is possible to draw general conclusions as to the approximate levels and directionality patterns of current jet engines. Although this information applies to a reasonably wide range of operating conditions for a few specific engines it cannot readily be scaled to predict levels on new or different engines. Such scaling as is possible (Figs.15(a),(b) and (c)) gives an accuracy of prediction which decreases rapidly as the differences in jet velocity and thrust of the two engines increase. A more accurate scaling procedure is not likely to be available until a better theoretical model of near-field noise pressures is available. Information on the noise levels forward of the jet and upstream of the intake is lacking at the moment but is likely to be required when the projected large rear engine installations of the supersonic transports are considered.

Whilst some information on rocket levels is available there are not enough unclassified data to enable generalized near-field levels to be obtained. Because the major noise source area appears to be some 20 diameters downstream the levels on the missile structure are lower than would be predicted by extrapolation from turbo-jet data.

As the noise is essentially broad-band with no discrete frequency components the data on spectra in the near field (Figs.17(a),(b)) are likely to be sufficiently accurate for structural response estimates. In the structurally important frequency range there is reasonable agreement between measured and predicted spectra.

The only full-scale correlation data available are from one engine[10] and these results are mainly for positions along the jet boundary. As this information is insufficient for the general case of the near field, more experimental measurements should be made on full-scale jets such as the Lockheed rig (Sec.1.2.13). Model experiments of the type being carried out at Southampton (Sec.1.2.3) should enable a better understanding of the correlation patterns to be obtained.

It is clear that the noise levels in practical configurations will differ from free-field predictions. Sufficient experimental data are not available to enable a reasonable assessment of the effects of ground and structural reflections, multi-jets, etc. to be made. There is a great need for such measurements on typical structural configurations.

SECTION II

SURFACE LEVELS AND SPECTRA IN BOUNDARY LAYERS*

2.1 INTRODUCTION

Much work is going on, mainly in the U.S.A., to investigate the nature of the pressure fluctuations at surfaces in a boundary-layer flow. The range of conditions is very wide, extending from underwater work (from small model experiments with rotating cylinders to tests on submerged submarines) on the one hand, to supersonic flight measurements and hypersonic wind tunnel wall experiments on the other. A considerable amount of data on r.m.s. pressure levels and spectra has been accumulated and further measurements are being made particularly in the high Mach number range. Only a few pressure correlation measurements have been made, primarily by Willmarth[34] and Harrison[35], but correlation techniques are being developed rapidly in the U.S.A. and it is to be expected that a large accumulation of data will be available in a year or so.

The majority of the investigations are confined to the measurement of pressure spectra at solid walls; some work is being done however to measure directly the response of a structure excited by boundary-layer pressure fluctuations (University of Minnesota and the R.A.E., Farnborough) and the noise produced by vibration (Ribner at U.T.I.A.) and to relate these to the excitation spectra measured by others. Theoretical approaches have been completed already using coincidence arguments by Ribner[36], Corcos and Liepmann[37], and Kraichnan[38-40] and more recently by Dyer[41].

2.2 EXPERIMENTS IN BOUNDARY LAYER NOISE

Experiments in the U.S.A. and Europe as known to the authors are listed in Table I. Clearly there may be further investigations not yet reported. Two basic sources of information exist, one the investigations carried for the Navy and obviously related to underwater noise, the other more general and related to aircraft and missile fatigue with less interest in externally radiated sound and more in internal noise in the environment of passengers, electronics, etc.

2.2.1 David Taylor Model Basin

Two types of experiment for spectrum measurements are reported, wind tunnel tests[35] and actual underwater measurements on a submarine. Some buoyant body measurements have also been made but no results are, as yet, forthcoming.

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*The major part of this section is based on the report *Boundary Layer Noise Research in the U.S.A. and Canada; A Critical Review*, by E.J. Richards, M.K. Bull and J.L. Willis[32] prepared after Professor Richards had visited the U.S.A. and Canada with the author in September 1959.

Harrison's early work (Fig.19) was done in an ordinary low-speed return-circuit wind tunnel driven by a fan and with presumably a fairly high residual noise level. Indeed, bearing in mind the poor results the authors have had in a similar tunnel owing to standing waves and a high noise threshold, the results must at first sight be suspect. Harrison claims, however, that the tunnel is unusually quiet and that no indications were obtained of such standing waves and high noise threshold. His greatest dissatisfaction lies in his instrumentation; the response characteristics of the Altec 21-BR-180 microphone used made it impossible to extend the investigation to frequencies above 2000 c.p.s. or to very low frequencies. Further work is going on at the David Taylor Model Basin to repeat this work and extend the frequency range in both directions. Harrison's overall root mean square pressure level equal to 0.0095 q is somewhat higher than the more established figure of 0.006 q given by Willmarth.

For correlation measurements an Altec Lansing microphone with a cap cover with a small hole in it is being used. By means of a system of rotating discs, the positions of the two holes in adjacent microphones can be varied easily from a few hundredths of an inch to four inches or so. Much interest has already been expressed in Harrison's rough figures of transverse correlations (Fig.20(a)). This new instrumentation will allow the repetition of this work with much greater accuracy. Filtered space correlations, shown in Figure 20(b), represent the only data available in filtered form at present.

2.2.2 Ordnance Research Laboratory, Pennsylvania State University

Three facilities have been used for noise level measurements. These are:-

(i) *Rotating Cylinder (18 in. diameter) in an anechoic water tank.* In these experiments hydrophones were mounted on the inside of the cylinder wall and outside the cylinder at some distance from the surface.

(ii) *The Garfield Thomas Water Tunnel.* Measurements were made in the boundary layers on the tunnel wall and on several streamlined bodies. In all cases the hydrophones were mounted flush with the wall under investigation.

(iii) *Buoyant Body.* This work is being carried out by the laboratory at Point West.

Buoyant body results are available but have not yet been studied. The rotating cylinder and water tunnel work has been reported[42,43].

No attempt has been made in these investigations to reduce the hydrophone size (½ in. to 5 in. diameter) to a minimum and, because of this, the results so far reported have not been generally accepted. However, Skudrzyk and Haddle (Ref.43, Part I) derive expressions giving the effect of hydrophone size on the measured spectrum and r.m.s. pressure, and use them as a basis for interpretation of experimental data (Ref.43, Part II).

These experiments are of particular interest in that they are aimed at investigating the effects of surface roughness. The influence of roughness on the frequency

spectrum is to increase the spectral levels at the high-frequency end, as seen from the water tunnel results in Figure 21. Skudrzyk interprets his experimental results as being consistent with the idea that roughness effects come into play only when the height of roughness exceeds the thickness of the laminar sub-layer in the boundary layer, in which circumstances small-scale turbulence generated by the roughness is responsible for the increase in spectral levels at high frequencies.

From spectrum measurements in the boundary layer of the Garfield Thomas water tunnel Skudrzyk deduces a value for the r.m.s. pressure of $2.4 \times 10^{-3} q$, somewhat less than half Willmarth's value.

2.2.3 California Institute of Technology

Willmarth's experiments (together with Harrison's at the David Taylor Model Basin) were some of the earliest carried out in the U.S.A., to determine r.m.s. pressures and pressure correlations. His first measurements of pressure levels and frequency spectra were reported in Reference 33. This work was later refined, and extended to include space-time correlations[34]. Willmarth, who is now at the University of Michigan, reiterated the results just referred to, and presented some further analysis of them at the recent Minneapolis Conference (W.A.D.C.), but has apparently not carried out further experiments.

From measurements with transducers of various sizes and extrapolation to zero transducer size, an overall r.m.s. pressure level of $0.006 q$ was obtained.

Willmarth's analysis of his space-time correlation measurements shows that the pressure field in the boundary layer is carried downstream at a velocity of 0.83 times the free-stream velocity, a figure substantiated by the work at the University of Minnesota, by Corcos at Berkeley, by Seraphini at Cleveland, and others. On the other hand, the lack of success accompanying his efforts at correcting the high-frequency end of the spectrum of pressure fluctuations on the basis of the finite size of the transducer using the assumption that the whole field of pressure is convected with speed 0.83U, has led him to suppose that the smaller scale pressure fluctuations either die out extremely quickly or must be convected at velocities considerably less than 0.83U. He urges the need for further experiments to determine the details of the relationship between the scale and convection speed of the pressure fluctuations.

2.2.4 Jet Propulsion Laboratory, Pasadena

Since the early work of Willmarth, interest in boundary layer noise at the California Institute of Technology has grown considerably, and possibly the most elegant work of all is being carried out at the Jet Propulsion Laboratory in Pasadena, an outstation of the Institute, financed by the N.A.S.A. A considerable addition to the data available, particularly in the field of high-speed boundary layers, comes from this laboratory. Here, Kistler and Laufer have measured r.m.s. pressures, frequency spectra, space-time correlations and turbulence convection speeds, in an 18 in. × 20 in. supersonic tunnel at Mach numbers between 1.6 and 5.

Good instrumentation is a feature of the laboratory; their hot-wire work in the wind tunnel, their development of very small crystal transducers and their correlation work all being of high order. In particular, their very small crystal transducer,

measuring only 0.030 inch in diameter allows considerable confidence to be expressed in the high-frequency end of the measured spectra, though the calibration falls off below about 8000 c.p.s. The ratio of r.m.s. pressure to dynamic pressure, $\sqrt{\overline{p^2}}/q$, in the Mach number range 2 to 5 appears to be somewhat lower than the subsonic value of 6×10^{-3}.

Valuable information on turbulence convection speeds and the rate of decay of the pressure correlation is contained in the space-time correlation results over a range of supersonic Mach numbers.

The derived speeds of convection of the pressure pattern for the supersonic Mach number range centres around 0.72U. In all cases the convection speed is supersonic. The space-time correlation figures were obtained using Willmarth's rather primitive method of integrating by means of a thermocouple, but this deficiency is unlikely to continue, a more accurate apparatus being now under construction.

2.2.5 Wright Aeronautical Development Center

Two types of full-scale experiments are reported by von Gierke[19] measurements of the noise levels, first of all inside the cockpit of an F.100 aircraft, and secondly at external microphones set flush at various stations along the fuselage of an F.102 aircraft. The Mach numbers covered the range from 0.40 to 1.16. In view of the complete lack in this country of practical data such as these, von Gierke's experiments, reported at the Minneapolis Conference and to be written up shortly by Eldred, are of vital significance. In Figure 22, for example, surface pressure fluctuations are plotted as overall levels against speed for three typical stations along the fuselage; values of the pressure levels in two frequency bands 20 - 75 c.p.s. and 1200 - 2400 c.p.s. are also included.

Von Gierke has plotted, (in Fig.23(a)), the information shown in Figure 22, together with additional data, as overall pressure levels against dynamic pressure (q) external to the boundary layer. He finds that an overall pressure level (re. 0.0002 dyne/cm^2) given by (83 + 20 log q) db (q in lb/ft^2) fits not only the flight experiments but also Columbia University's results for rotating cylinders. von Gierke also includes results from a microphone inside the cockpit of the F.100 and finds that they also fit on the same straight line after correction for wall attenuation. Little confidence can be placed in these measurements because the correction for wall attenuation and internal absorption cannot be made with any appreciable accuracy. It may be noted that the above expression for overall pressure level is equivalent in more familiar terms to $\sqrt{(\overline{p^2})} = 0.006\,q$, i.e., Willmarth's figure.

Von Gierke is an adept at non-dimensional collapsing of experimental results. He has done this in Figure 23(b) for the pressure spectra obtained in the experiments referred to above, and obtains a surprisingly good collapse of the data available to him by plotting $10 \log_{10}\{P(f)UM/\overline{p^2}\delta\}$ against $f\delta/MU$ (instead of the usual $10 \log_{10}\{P(f)U/\overline{p^2}\delta\}$ against $f\delta/U$ (see Ref.35). The innovation here is the inclusion of Mach number in the non-dimensional parameters, although it is difficult to see why Mach number should enter in this particular way. Indeed, von Gierke makes no pretence at a theoretical justification for including Mach number in this way, and claims only that 'it works' for the available data.

2.2.6 Columbia University

Rotating-cylinder experiments have been carried out at Columbia University by W.L. Nelson[45]. This paper has not yet become available to the present authors but the results are included with the work of von Gierke (Figs.23(a) and 23(b)), who used them to extend his own curves to much higher dimensionless frequency parameters. It is the collapsing of these results, covering a wide Mach number range, into a single curve, by using the von Gierke parameter which is most interesting and deserving of further study.

2.2.7 N.A.S.A., Cleveland

This laboratory has an excellent boundary-layer facility in the form of a 40 ft long channel of cross-sectional shape, 24 in. x 9 in., all but the working wall being relieved of boundary layer by suction. Stray noise originating downstream of the working section is excluded from the working section by a sonic choke similar to that used in this country. Mach numbers of around 0.6 are standard, though an M = 2.0 stream is obtainable. Unfortunately, the results of work going on at the Lewis Laboratory are being held back until Seraphini submits them in support of his Doctor's thesis at Case University. The results, however, seem to agree with others, particularly in so far as turbulence convection speeds are concerned.

In the same laboratory, L.W. Baldwin has obtained similar data in the fully developed flow in a circular pipe. Baldwin's Ph.D. thesis[46] describes this work.

2.2.8 Institute of Aerophysics, University of Toronto

The experiments being carried out here are to excite, by boundary-layer pressure fluctuations, a thin panel let into the wall of a quiet wind tunnel, and to measure the spectrum of the noise occurring in a reverberation chamber surrounding the tunnel. No results of this work are available as yet, although they promise to be of considerable interest.

Wilson has also carried out investigations of the effect of surface roughness on a rotating cylinder[47].

2.2.9 University of Minnesota

Some experiments in small ducts are proceeding in the Electrical Engineering Department under the guidance of Lambert. Correlation of pressures in space and time are being made and convection speeds in agreement with other investigators are being computed. As the first step towards investigating structural response, measurements are being made on a rubber membrane inserted in the wall of a wind tunnel.

2.2.10 Edwards Air Force Base

Flight measurements have been made of r.m.s. pressures and frequency spectra with internal and external microphones at various stations along the fuselage of a B-47A aircraft[48]. Internal noise measurements were also made for the D-558-II research aircraft.

2.2.11 College of Aeronautics, Cranfield

Lilley and Hodgson[49] are carrying out theoretical and experimental studies of the surface pressure fluctuation on a wall jet. In this installation a 1.5 inch diameter air jet impinges normally onto a 2 inch thick wall situated 0.75 inch away from the jet orifice. The working range of wall velocities is 40 - 140 ft/sec. Preliminary results give values of $\sqrt{(\overline{p^2}/q)}$ of the order of 0.1, i.e. about 17 times as great as the generally accepted figure for corresponding measurements on a flat plate with zero pressure gradient. The increase is thought to be due to the greater thickness of the shear layer and the relatively high intensity of turbulence although it is difficult to see how the jet noise can be eliminated in the frequency range of measurement (40 - 20,000 c.p.s.). Auto and space-time correlations are also being made to enable equivalent eddy convection speeds to be established together with the detailed turbulent structure.

2.2.12 Southampton University

The wall of a 6 in. x 2½ in. wind tunnel has been used to obtain pressure levels and spectra at Mach numbers of 0.5 and 1.2 to 1.54. Preliminary results are shown plotted on Figure 29. Space-time correlation measurements of wall pressures have also been made, and filtered, space correlation results show good agreement with Harrison's curve (Fig.20(b)). The aim of the work is to accumulate sufficient data to allow a realistic empirical formulation of the pressure field to be made for use in structural response calculations and also to find out how various regions of the boundary layer contribute to the wall pressure field. A 9 in. x 6 in. tunnel which will be suitable for studying the vibration of a panel fixed in the wall is under construction.

2.2.13 R.A.E., Farnborough

Interest is mainly in the surface pressure fluctuations and structural response of an aircraft flying in the Mach number range 1 to 3. In a preliminary experiment a rocket-boosted model was used and measurements made on a wing surface in the free-flight stage. The wing was of double wedge cross-section 5% thick having a 19 in. chord and a 9 in. semi-span. At velocities in the region of 2000 ft/sec wing surface pressure fluctuations on the rear wedge were measured by a flush mounted microphone. Provision was also made for the study of the vibration of a 6 in. by 4 in. panel let into the rear wedge.

A more extensive programme has been carried out on the Fairey Delta 2 aircraft[51] flying over various flight paths at Mach numbers extending up to 1.6. A section of the skin of the lower surface of the port wing was replaced by a thick rectangular plate. Towards one end of this plate a microphone was mounted flush with the external surface and towards the other end a test panel was formed by reducing the plate in thickness to 0.018 in. over an area 6½ in. by 8½ in. by chemical etching. Panel strains were measured at the centre and close to the edge by barium titanate strain gauges. The overall accuracy of the pressure measurements is stated to be ± 1½ db.

The pressure measurements indicate that over this range of speeds and Mach numbers the overall levels increase linearly with the dynamic head q as shown in Figure 24. The spectrum is flat up to the working limit of the instrumentation at 10 Kc/sec and gives the overall level $\sqrt{(\overline{p^2})} = 0.0026\,q$. Measurements of the boundary layer give

displacement thicknesses which suggest that the spectrum should continue flat up to about 26 Kc/sec. If allowance is made for this additional energy in the unmeasured portion of the spectrum then the value for $\surd(\overline{p^2})$ becomes very close to Willmarth's figure of 0.006 q.

The variation in overall level with Mach number at constant E.A.S. is shown in Figure 25. From this graph it can be seen that both the pressure and strain follow closely the E.A.S. curve. A plot of all the results against Mach number is shown in Figure 26. Here there is considerable scatter (± 1½ db) but no general trend of reduction with Mach number is apparent. Measurements of the boundary layer showed that the thickness remained substantially constant for all the flight conditions of interest.

Spectra of strain in the test panel are shown in Figure 27 for three Mach numbers during the flight shown in Figure 25. There are peaks in the spectra at about 390 c.p.s. which are associated with a mode, identified on the ground by loudspeaker tests, as having a node line across the centre of the panel normal to the flow direction. The fundamental mode at 200 c.p.s. was not excited by the boundary layer pressure fluctuations.

An interesting comparison was made between jet and boundary layer excitation. The panel was subjected to noise from a jet engine at the same overall noise level and a comparison of the strain responses is shown in Figure 28. For the same overall level of excitation the response to boundary layer noise is only about 20% of the response to jet noise. In the jet noise case the fundamental mode of vibration was excited predominantly. The difference in the response levels was partly due to the fact that the excitation pressure energy in the frequency range of major structural response is less in the boundary-layer case than in the jet case for equal overall excitation levels. The different correlation patterns must also be playing an important part, as in the jet-noise case the fundamental mode is predominant whereas in the boundary-layer case the first harmonic is predominant.

Further measurements to cover a wider range of Mach numbers and boundary-layer thicknesses are being considered.

2.3 THEORETICAL WORK ON NOISE DUE TO STRUCTURAL VIBRATIONS

Up to the present, experimental effort has been mainly concentrated on building up knowledge of the statistical properties of the pressure fluctuations in boundary-layer turbulence; little experimental work has been done on structural response and the noise field generated by the vibrating structure, although some is now in progress at U.T.I.A. and Minnesota University.

Theoretical methods have been put forward by Ribner, Corcos and Leipmann, Dyer, and Kraichnan, but because of the lack of experimental information on the statistical properties of the turbulence, it has been necessary to assume physically reasonable expressions for the correlation functions in order to obtain any quantitative results.

Other work having some relevance to the problem is given in References 52, 53, 54 and 55.

2.3.1 H.S. Ribner

Ribner's paper[36] treats in detail the idealised case of convection of a rigid pattern of turbulence, i.e. that the frequencies of pressure fluctuations in a turbulent eddy are very much less than U_c/l where U_c is the convection velocity and l the panel length. The frequency at a fixed point due to convection of wave number K is then $KU_c/2\pi$. However, the basic approach can be generalized to allow for fluctuations of the pattern. Such a generalization is now being explored.

The major contributions to the panel response come from coincidence of wave number and speed in the turbulence pattern with the wave number and speed of free flexural waves in the panel (which is treated as infinite). By assuming the form of the turbulence pressure correlation and by making simplifying acoustic assumptions to obtain noise levels due to reverberant build-up inside a cylinder, Ribner obtaines expressions for noise levels in an aircraft fuselage. The results obtained are that, for thin boundary layers, the noise level varies as $U^5\delta^3$ (U = aircraft speed, δ = boundary-layer thickness) and as $U^3\delta$ for thick boundary layers. Present U.T.I.A. experiments are aimed at establishing the validity of the $U^3\delta$ and $U^5\delta^3$ laws. Preliminary results exhibit an accurate U^5 law.

It may be noted that there is an inconsistency between the assumed form of space correlation and the spectrum function which Ribner uses in the development of the theory. Nevertheless the spectrum function appears to be physically reasonable. As it stands, Ribner's work is applicable to the case in which acoustic damping of the structural vibration may be neglected, but could be extended to the submarine case by taking proper account of acoustic damping, which in water may be many times greater than the structural damping itself.

2.3.2 Corcos and Liepmann

Corcos and Liepmann[37] whilst allowing for pattern fluctuations make the restrictive assumption that there is negligible correlation between motions of surface elements. This is equivalent to assuming that the correlation of the plate vertical acceleration is essentially a delta function. The structural response is again obtained from coincidence arguments. No attempt is made to assume a definite functional form for the boundary-layer pressure correlations and results are given in terms of the unknown correlation functions.

The analysis yields the relevant similarity parameters, but there will obviously be considerable scope for further application to prediction of noise intensities, when the required experimental data become available.

2.3.3 I. Dyer

Dyer[41], like Ribner, assumes the form of the pressure correlation of the turbulence, but proposes an improvement on the rigid convection pattern approach by including the effect of the finite statistical lifetime of the turbulence, i.e. a correlation function of the form:

$$\exp[-K\sqrt{(\zeta - U_c\tau)^2 - \xi^2} - |\tau|/\theta]$$

(ζ, ξ = longitudinal and transverse separation distances, τ = time delay, θ = statistical lifetime of the pressure fluctuations), and on the basis of experiment assigns the values $K\delta^* = 2$, $\theta U/\delta^* = 30$, $U_c = 0.8U$. However, for mathematical simplicity Dyer reduces this to a delta function in space.

The frequency spectrum is then obtained in terms of a characteristic frequency, ω_0 which is given by $\omega_0 = KU_c + 1/\theta$ and consistent with the above figures $\omega_0 \approx KU_c$. Justification for these values is given by Figures 20(a) and 31.

The structural response is considered for two cases, viz.,

(a) For the coincidence condition (turbulence wavelength equal to plate modal wavelength convected at the speed which gives a frequency equal to the mode frequency) where the convection speed is of the order of the speed of free flexural waves in a plate, and

(b) For convection speeds below coincidence speed.

These two cases are appropriate to aircraft and underwater applications respectively.

As would be expected, the analysis shows that at coincidence (and with low damping) the structural response is increased by an increase in turbulence lifetime, and by decreasing damping, so that an increase in damping is an effective means of reducing noise.

In the underwater case there are two important restrictions on the use of increased damping as a noise reducer. The first arises as a result of liquid coupling so that increasing hysteretic damping is not an efficient means of noise reduction unless

$$\eta > \frac{\beta_1}{\omega_{mn} M'}$$

where η = hysteretic damping coefficient
β = viscous damping due to liquid
M' = effective mass per unit area of vibrating plate
ω_{mn} = mode frequency.

The second is that damping is effective only at frequencies below a transition frequency, given by $1/\pi\eta\theta$, where θ is the statistical lifetime of the turbulence. At frequencies greater than this, damping becomes less and less effective in reducing structural response. It might be noted that in this case hysteretic damping and the decay of turbulence play analogous parts in the plate response.

2.3.4. R.H. Kraichnan

Kraichnan[38-40] obtained a representation of the spectrum of turbulent pressure fluctuations from considerations of longitudinal and transverse pressure dipoles moving with convection velocities distributed about a mean. He then considers the response of finite rectangular plates to this excitation, and obtains expressions for the sound radiation, due to plate vibration, in free space on the side of the plate

remote from the flow. Results are given for the longitudinal and transverse dipole contributions to the sound radiation for the cases of critical and very low damping of the plate vibration modes. In general the effect of transverse dipoles is not significant in comparison with longitudinal dipole effects.

For moderate subsonic Mach numbers and thin boundary layers, Kraichnan concludes that the total transmitted power varies as $U^5\delta^4$, and that the dominant transmitted frequencies increase with Mach number.

2.3.5 Bolt, Beranek and Newman

An approximate method of predicting the level of noise due to the boundary layer is contained in a Bolt, Beranek and Newman report[23] concerned with estimation of noise levels in aircraft and missiles generally. The essential steps in the procedure are:

(i) Estimation of octave band pressure levels on the outside of the fuselage,

(ii) Estimation of transmission loss for an isolated panel,

and (iii) Correction for characteristics of the receiving space.

External pressure levels are based on a representative boundary-layer pressure spectrum obtained from noise measurements in aircraft, corrected for fuselage transmission losses. Transmission loss is estimated in a similar fashion to that for air-borne noise, the frequency range is divided into regions where one effect (mass, stiffness, panel resonance, etc.) predominates and methods are given for estimating the transmission loss for each, so that a composite transmission loss-frequency curve can be built up.

A fundamental treatment of the various phenomena is not given, the intention of the method being to provide preliminary values of noise levels for design purposes.

2.4 DISCUSSION

The first impression to be gained from the above catalogue of boundary layer noise results is that of the wealth of experimental results now available and the lack of necessity for further similar work. A further study of the situation, however, tends to show how little we know of the phenomena involved and the enormous scatter of results between the various experiments. At some frequencies, r.m.s. pressures differ by factors as high as 30 or more, a serious matter in regard to the prediction of fatigue life, though of less importance possibly in predicting internal noise levels. Much of the discrepancy arises, no doubt, from experimental difficulties, but the greater part is real and emphasises the need for a carefully planned and carefully instrumented series of experiments to relate wall pressures to turbulence.

2.4.1 Root Mean Square Pressure Levels

Of the statistical parameters associated with turbulent pressure fluctuations, which are required for prediction of structural stress levels and internal noise levels, the r.m.s. pressure is the one most firmly established. Although values of

$\{\sqrt{(\overline{p^2})}/q\} \times 10^3$ at subsonic speeds of 1.3 (Mull and Algranti[50]), 3 - 5 (McLeod and Jordan[48]), 4.5 (Callaghan[56]) and 9.5 (Harrison[35]) have been measured, *the most generally accepted value is 6* as measured by Willmarth and others (Sec.1.2.3).

2.4.2 Frequency Spectrum of the Wall Pressures

Although wall pressure spectra have been measured by many investigators, the stage has not yet been reached where it is possible to predict the spectrum very accurately for any given set of conditions.

The various experimental data are compared in Figure 29. Results obtained by the authors in the 6 in. × 2½ in. subsonic wind tunnel and 2 in. diameter water tunnel at Southampton are included, in addition to those from American sources. It will be seen that there is a very wide scatter of results, which is most pronounced at the high-frequency end of the spectrum. The spectral density tends to be constant up to some cut-off frequency and to fall off with frequency above this. Theoretical work also indicates a spectrum of this general form. Willmarth's results show that the high-frequency attenuation increases with the ratio of transducer size to boundary-layer displacement thickness (d/δ^*) .

Mach number does not appear to have a marked influence on the frequency spectrum at subsonic speeds; at supersonic speeds an increase in Mach number may be accompanied by a fall in non-dimensional spectral level and an increase in the non-dimensional cut-off frequency, by a factor of about the same mangitude as the ratio of the Mach numbers involved. Thus if we followed von Gierke's practice of plotting against $f\delta^*/UM$ better agreement between the supersonic curves would be obtained. This had been done but, as might have been expected, the lower the Mach number, the more the subsonic curves are displaced towards higher dimensionless frequencies. It therefore appears that von Gierke's parameters do not adequately account for Mach number effects.

2.4.3 Longitudinal Space-Time Correlations and Convection Velocities

Convection Velocities. Longitudinal space-time correlations so far measured have been used mainly for the determination of convection speeds of the pressure pattern as a whole. Of considerably more interest is the distribution of convection speed over the space wave lengths of the turbulence. From the effects of transducer size on the frequency spectrum, Willmarth has been led to the conclusion that small-scale pressure fluctuations die out very rapidly or are convected at speeds significantly less than the overall convection speed.

The relation between convection speed and frequency can be obtained by filtering the signals into narrow bands before correlating, but this technique will not distinguish between fast moving large eddies and slow moving small eddies. Such a method was applied by Harrison[35] who found that, for frequencies below $f\delta^*/U$ of the order of 0.3, the convection velocity was constant at about 0.8 U. Since the frequencies covered by Willmarth's experiments correspond to $f\delta^*/U$ less than 0.3 there is a suggestion here that the variation of rate of decay with frequency may be more important than the variation of convection speed. However, some evidence in support of Willmarth's conclusion can be obtained from work on jets. That there is a considerable variation of convection velocity with frequency in a jet is shown by some

preliminary results obtained at Southampton by Williams and Barrett. This work is to be reported at a later date. Convection speeds have been obtained at various positions in the jet from hot-wire measurements and these are shown in Figure 30. In the results for a boundary layer, being the inverse of those for a jet, we would expect the convection velocity in a boundary layer to decrease with frequency. Obviously a more detailed experimental investigation is required to clarify the situation regarding the relation between frequency and convection speed.

The auto-correlation of pressure fluctuations (at a point moving with the mean convection velocity of the turbulence) can be obtained from the envelope of the space-time correlations. Such values for subsonic speeds are shown in Figure 31. Also shown is the exponential correlation function assumed by Dyer in this theoretical work, corresponding to $U\theta/\delta^* = 30$. This gives a reasonable representation of the experimental data for $\tau U/\delta^*$ up to about 70, but the experimental correlation does not decay so rapidly at larger time delays.

At supersonic speeds the lifetime of pressure fluctuations appears to be less than for subsonic speeds by a factor of about 10. This is the one striking difference between subsonic and supersonic pressure fluctuations revealed by the experimental data so far.

Measurements of space-time correlations have been made at fairly widely separated points (in terms of δ^*) so that they do not yield very much information on instantaneous space correlations. In particular, for the space correlation of the pressure field at zero time delay (given by the intercepts on the vertical axis of the correlation curves for constant spatial separation) we have no values of the correlation coefficient at separations between those for which it is unity and almost zero. The results we have emphasise the difficulty in increasing the amount of experimental information here, for they show that this space correlation falls off to zero over a distance of only 2 or 3 boundary-layer displacement thicknesses (i.e., over a distance of the order of 0.2 inch for the scale of most laboratory experiments). Willmarth has derived this function from measured frequency spectra and overall convection speeds (see Fig.20(a)) and has observed that the comparison with Harrison's transverse space correlations indicates a marked anistropy in the structure of the pressure pattern.

2.4.4 Conclusion

The above discussion indicates that while the experiments which have been carried out have added considerably to our knowledge of pressure fluctuations, and allowed some general conclusions to be reached regarding the likelihood of acoustic fatigue due to boundary-layer noise, the inaccuracies of instrumentation and clear inconsistencies observed between the various experiments suggest the need for much further work of a more controlled nature. For example, the magnitudes of the pressure fluctuations are probably accurate enough to suggest that there is no likelihood of acoustic fatigue of structures in the envisaged range of indicated air speeds and Mach numbers; at the same time they are of such a magnitude as to suggest that internal noise in supersonic transport aircraft can be very high and the need is emphasised for a much better theoretical method of calculating the structural response and the consequent internal noise pattern in the cabin.

For such purposes, it is still necessary to set up empirical expressions for the space-time pressure correlation over the surface of the structure to represent analytically the essentials of the physical system; available data allow us to put only rough quantitative values to the parameters involved. This procedure has been followed by Dyer (see Sec.1.3.3), but the representation is by no means precise. There is still a great need to extend correlation work to produce sufficient data of the type already obtained, in order to define the statistical properties of the pressure field more precisely.

SECTION III

STRUCTURAL RESPONSE

3.1 INTRODUCTION

The majority of aircraft companies and research establishments have now been faced with the problem of structural failure due to jet noise and a wide variety of design and test procedures have been adopted. In several cases the onset of incidents resulted in 'crash' programmes which inevitably involved considerable work and expense without resulting in any general method.

A considerable amount of work has now been done on all the various aspects of the problem but it is still not possible to pick out a standardised method of approach which can be guaranteed to give consistent results. It is the purpose of this review, therefore, to attempt to sift out the information of general value and to indicate which items need further work. Several companies have now adopted their own semi-empirical procedure but there is wide variation in these methods.

The theoretical work of Powell[57,58] and others presents a reasonable working framework for the study of the relevant parameters. However, it is not yet possible to apply this directly to actual structures because the normal modes of stiffened structures are not known accurately. Studies of damping appear to be close to the stage where the basic structural damping could be estimated if the mode shapes were known. The question of the damping due to sound radiation is not finally solved but work is progressing in this field.

Owing to the lack of a working practical theory of structural response there is a wide variation in test facilities. In the U.S.A. and Canada the majority of practical designs are developed in some form of siren facility. The degree of confidence placed in siren tests varies such that in some cases no further testing is done whereas in others a large section of structure is tested behind a jet engine. In Europe sirens have not been used in the design of any aircraft and in fact there is no siren of the higher intensity type yet in operation although a lower-intensity random siren facility has just been completed at Southampton University, and a higher-intensity discrete-frequency siren is in the course of construction at Vickers Supermarine. Development in Europe has generally been carried out with a jet engine as the noise source.

Little success has been achieved in trying to determine relative merits of structural configurations. This now seems inevitable as detail design practice varies considerably.

3.2 EXPERIMENTAL AND THEORETICAL WORK

A mass of test work has been carried out on structures ranging from full-scale aircraft down to simple cantilever specimens. There has also been a range of methods of load simulation varying from actual engines to discrete-frequency sirens and

vibrators. It is not possible therefore to give full details of all the tests which have been carried out but an attempt is made to refer in greater detail to those which add significantly to our knowledge of the problem whilst mentioning only briefly those tests which are a repetition of earlier work.

3.2.1 R.A.E., Farnborough

Two full-scale jet engines (one with an afterburner) on an open air site have been available at the R.A.E. for structural response and fatigue tests. Sections of a missile skirt structure were tested close to the engine with afterburner to simulate the rocket environment, and it was confirmed that the spot welding of stiffeners to conventional skin was unsatisfactory. A similar structure with rivets in place of spot welds proved satisfactory.

A programme of proof tests on a series of tailplane test specimens was used to obtain strain levels and spectra. The distribution of the strain amplitude was checked in some detail and was found to approximate closely to Gaussian.

Work on structural response due to boundary-layer pressure fluctuations has been quoted in Section 2.2.13.

3.2.2 De Havilland

Work has been aimed at developing satisfactory detail design for the Comet IV rear fuselage and tailplane, which are situated in areas of overall noise levels of about 150 db. A section of the tailplane of 5 ft span and full chord was tested behind a jet engine on the open air rig at the R.A.E. In order to fix the position of the test specimen strains were measured on the rear fuselage and tailplane of the prototype aircraft operating with and without the Rolls-Royce suppressors. The position of the test section was then adjusted behind the Ghost engine on the R.A.E. rig in order to give approximately the correct strain levels in the skin. In this position (3.5 diameters downstream and 1.5 diameters out from the jet boundary) the overall noise level was 151 db. Strains were measured in the centre of panels, on skin over stringers at rib intersections and in the rib flange bend radii where failures had previously occurred. It was found that the effect of the suppressors was to give a 20% reduction in the tailplane skin stress. The first three specimens were used for fatigue tests and the fourth one is being used for more detail strain measurements and investigations of the mode shapes (see Sec.3.2.5).

In parallel with these larger-scale tests, small-scale fatigue tests on the skin-stringer to rib connection are being conducted[59]. A mid-bay to mid-bay section of skin with one top hat stringer is attached to a section of the rib which is mounted on a vibrator. The fundamental free-free mode of the skin and stringer section is excited to give maximum skin stresses above the rib attachment.

Skin stress levels have been measured at critical skin/rib intersections on the aircraft and then the test is run at similar skin surface stresses. The work has then been extended with tests at higher and lower stress levels to provide S-N curves for such skin rivet lines.

A typical test specimen is shown in Figure 32 together with the measured stress distribution in the region of the rivet line. A typical set of results is shown in Figure 33 for DTD.710 material. The S-N curves show considerable scatter on the individual points but the general trend is to show a very flat curve in the region of 10^6 to 10^{10} reversals. Tests using DTD.746 material showed no significant difference in life at these low stress levels. The plain holes gave approximately a tenfold increase in life over spin dimpled holes in the lower stress range.

In this type of test the significance of the test in terms of aircraft life depends critically on how accurately the stress distribution in the cross section at the rivet line is reproduced. If there is any possibility of membrane stresses in the skin of the actual structure then the aircraft life may only be about half that predicted from the tests.

3.2.3 Vickers

A proof test on a tailplane specimen for the V.C.10 was carried out on the noise rig at the R.A.E. No strain measurements were made. The overall levels were only of the order of 146 db, the tailplane skin is 19 s.w.g., and the elevators are of solid honeycomb. No serious difficulty with this type of structure in these noise levels should be expected. Several design details are to be tested in the Southampton University siren facility.

A discrete-frequency siren with output of 165 db over the frequency range 100 c.p.s. - 2000 c.p.s. has been designed. The mode of operation is to mount the test panels, 3 ft x 3 ft, at the mouth of a hypex horn to give a normal incidence configuration.

The siren is being designed to be operated by an 80 lb/in.^{2} air supply. The rotor is 9½ in. O.D. and has 20 ports. Drive is from a 3 H.P. motor.

The test chamber consists of two enclosures one inside the other. The inner chamber, 9 ft x 6 ft, has concrete wall 4 in. thick and there is an absorbent lining of 4 in. rockwool leaving a 4 in. air cavity between the rockwool and concrete. Cooling air is provided to prevent the rockwool from heating up. The outer chamber (an old air raid shelter) is also lined with a 4 in. layer of rockwool.

A speed control for the siren motor is being considered which would enable the siren frequency to change as the resonance frequency of the panel changed due to fatigue cracks developing, and so maintain excitation at resonance. As the damping of the panels is likely to be very low the frequency-response curve of the panel will be very peaked. It is doubtful therefore whether the proposed systems would be sufficiently stable to ensure that the siren frequency was also maintained at the resonance frequency of the panel.

3.2.4 Imperial College

Attention has been centred on the effect of a backed damping layer on the vibration of a simple steel or aluminium strip[60]. The fundamental properties and mode of operation of the damping layer are being studied. The material being used is the Johnson and Johnson 'Permacell', which consists of a fabric impregnated with a latex-

based adhesive and having an aluminium foil backing. The original programme set out to verify experimentally the work of Ross, Kerwin and Dyer[61].

Test strips are 3 ft long by 1 in. wide and $\frac{1}{8}$ in. thick for aluminium or 1/16 in. thick for steel, with a similar test arrangement to that referred to in Reference 61. The bar is supported vertically by a long thin wire, it is excited at one end in a 'free-free' bending mode and there is a capacity pick-up at the other end to measure the response. In this case the damping was measured by the free-decay method which limits the validity to cases of low damping.

The general trend of the B.B.N. theory is verified for this material but results on aluminium and steel show that it is not good enough to assume that the fabric reinforced layers behave as a homogeneous material of equivalent properties. The damping of the aluminium strips is 5 times as great than would be estimated from the damping of the steel using the homogeneous assumption. The material itself is found to have a Young's Modulus of the order of 100 lb/in.^{2}.

The theoretical variation of damping with the shear parameter (or frequency for a specified bar) is borne out only when the damping material has a perfectly flat backing layer. When the foil backing layer is wrinkled as in the production article the damping is increased at the lower frequencies. This condition can also be obtained by cutting slots in the backing layer and the optimum spacing is such that distance between slots is approximately equal to the shear diffusion length l. The maximum damping is not increased by these slots but the frequency range is extended.

There has been no investigation of the effects of temperature and humidity.

3.2.5 Southampton University

3.2.5.1 Experimental Study of Random Vibrations

An experimental investigation is being carried out on full-scale structures in collaboration with Sud Aviation and the R.A.E. to determine the types of modes of vibration which are being excited in typical aircraft structures by jet noise. The first results relating to the Caravelle rear fuselage, have now been reported[62,63]. Recordings were made of the strains induced in the rear-fuselage test structure when one engine was running at maximum take-off thrust. The structure is conventional sheet/stringer combination attached to pressed out frames.

The analysis to date has been concentrated on the strains in the centres of the panels. A typical strain spectrum is shown in Figure 34. Correlation measurements have indicated that the lower frequencies having low strain amplitude are associated with overall modes of vibration. The larger panel strains occur at higher frequencies with the frames acting as boundaries. In these measurements the main resonance peak in each panel occurs in the range 600 - 700 c.p.s. and has been identified with the fundamental stringer twisting mode (i.e. adjacent panels out of phase). There are generally two smaller peaks in the 800 - 1000 c.p.s. range but the corresponding modes of vibration have not been completely identified due to lack of sufficiently closely spaced strain gauges. A typical correlation spectrum for two adjacent panels is shown in Figure 35. This form of analysis presents clearly the phase relationships which exist at each predominant frequency and hence determine the mode shape.

An attempt has been made to calculate the panel resonant frequencies theoretically on the assumption that the frames act as boundaries. Lin's theory (see Sec.3.2.16) and also an energy method have been used with some success. The energy method is being checked on a simplified flat sheet/stringer frame structure having similar relative stiffnesses to the Caravelle fuselage. A curved specimen is being constructed to check extensions of the theory.

It should be emphasised that although these results are very encouraging they apply only to the type of structure investigated. At the R.A.E. a similar programme is being carried out on a tailplane specimen whose structure is essentially thick skin, torsionally stiff stringers and more flexible ribs. Strain spectra (Fig.36) have been obtained and the predominant skin response mode has been identified[64].

3.2.5.2 *Normal Modes of a Stiffened Cylinder*

A theoretical study of the modes of vibration of a stiffened cylinder is being undertaken. The characteristic equation giving the natural frequencies of vibration of a circular fuselage with stiffeners of arbitrary but uniform pitch has been formulated. Current work is studying the effect of stringer cross-sectional distortion.

3.2.5.3 *Response of Continuous Linear Systems to Random Loading*

A theoretical study of the use of wave theory to solve vibration problems of continuous linear systems is being made. An influence function theory for the vibration of finite plates and shallow shells is being developed. The basic influence functions corresponding to the forced vibrations of unbounded plates and shells have been derived. It has been found possible to extend this work to finite plates by superimposing two infinite plate solutions, one representing the loading and the other the conditions at the boundary. The application of the theory to the analysis of random excitation of circular plates and shells seems possible and is being studied.

3.2.5.4 *Damping*

The various contributory factors to the total damping of a typical aircraft structure are being considered separately. Measurements of the energy dissipation at a riveted joint in a beam vibrating in a free-free mode have been completed. The investigation was carried out for peak rivet loads in the range from 0.1 to 200 lb. Typical curves for two frequency regions are shown in Figure 37 where energy dissipation per cycle/(rivet load)^2 is plotted against rivet load. The damping is almost linear up to a rivet load of about 10 lb. The onset of slip is accompanied by a large increase in damping. When the joint is subjected to continuous harmonic excitation above this critical load the damping drops steadily with time and the critical load increases. Theoretical work on the energy dissipated in a visco-elastic interface between the two plates of a riveted joint has been completed[65]. To verify this theory experimentally an apparatus has been built to measure the stiffness and energy dissipation of a joint with a visco-elastic interface. A rig for measuring the dynamic shear modulus of the visco-elastic material over a wide range of temperatures has also been constructed. As the increased flexibility of the joint may lead to rivet fatigue failures a fatigue test rig has been built to test the various joint configurations.

The damping of a structure due to acoustic radiation has also been investigated theoretically and experimentally. The acoustic radiation damping of a rectangular panel with either simply-supported or fully-fixed edge conditions is shown in Figure 38 for the fundamental mode of vibration. Measurements made on a simply-supported panel in a cylindrical baffle support the theoretical predictions. In the experimental work the panel was also tested in a vacuum chamber to measure the inherent structural damping present. Further experimental work is planned to investigate panels with different length/breadth ratios and also panel groups.

The inherent structural damping of honeycomb panels is being studied experimentally and theoretically[66]. Preliminary results indicate that the basic material damping of honeycomb skins is of the order of 3 times greater than plain alloy sheet. A resonance tube, 2 ft diameter, is being constructed to provide a high-intensity noise field for the fatigue testing of honeycomb structures. A sliding horn in front of the panel and piston behind the panel will enable the tube to be tuned to the panel resonance.

Considerable effort has also been put into the study of the use of an added damping compound in reducing random vibrations. Basic studies of a coating of a high polymer, vermiculite filled material have been made and its effect in damping out beam vibrations measured. This type of damping material developed by Oberst[67] has appreciable stiffness and acts most efficiently when working in direct stress. Because of the increase in stiffness due to the damping layer the natural frequencies of coated beams are appreciably higher than those for uncoated beams. The optimum distribution of this type of material for given mode shapes has been studied and it has been confirmed that it is most effective when applied to regions of maximum direct strain amplitude. This is in contrast to the soft damping materials used in damping tapes which dissipate energy in shear and add little to the stiffness of the structure.

Thus with the exception of the basic material damping (which is low compared with other sources) most of the sources of damping are being studied and methods of increasing the damping of a structure are being investigated. A theoretical study of the criteria for comparing the effectiveness of different damping configurations has recently been reported[68]. In this work expressions have been derived for the response of simple systems treated with damping materials and excited by randomly and harmonically varying forces. It is shown that a treatment which increases stiffness decreases vibration amplitudes but increases inertia forces, whereas the increase in the mass and damping of the system is always beneficial.

3.2.5.5 High Intensity Test Facility

A siren test facility has been constructed with the aim of evaluating testing techniques. The installation is of the grazing incidence type where the specimen is placed at right angles to the wavefronts emerging from the exponential horn fitted to the siren. The size of specimen is 2 ft × 3 ft - the 3-ft dimension being parallel to the direction of propagation of the sound wave. The siren is of the broad-band type having a band width of 130 c.p.s. (horn cut off) to 1000 c.p.s. (exciter cut off). A vibration generator actuates the nozzle gate mechanism and in this way any shape of spectrum in the range 130 - 1000 c.p.s. can be obtained by feeding the appropriate signal to the vibrator. It is planned to compare discrete-frequency testing with broad-band testing on a representative sheet/stringer frame structure and also to

carry out detail noise calibration of the levels, spectra and correlation obtainable on the specimens. The maximum level on the test specimen is 150 db.

3.2.6 O.N.E.R.A.

The programme consisted of testing panels representative of Caravelle sizes (380 x 80 mm) behind a Mabore jet engine. Twelve panels were mounted in a frame all the way round the jet at a position 1 metre downstream where the noise level is equal to the Caravelle level. The panels were fully fixed along the edges with great care being taken to reproduce the fixing condition. All clamping bolts were tightened by torque spanners. Brittle lacquer coatings were used initially to ascertain the positions of maximum strain and thus the appropriate positions for strain gauges.

It was found that with the engine used in these tests the noise pressure, which had a peak octave level of 130 db in the 300 - 600 c.p.s. octave, did not become steady until after the engine had been running for 5 minutes. Initially there was a drop in level until after 5 minutes a steady value, 5 db lower than the initial peak, was reached.

Preliminary test results varied considerably for nominally identical test panels and, therefore, a rigorous control of preparation of the specimen was introduced. The plates all came from the same billet, passed through the same rollers and had the same treatment.

The effect of damping the panels with applied surface coatings was investigated. It was found that a $\frac{3}{8}$ in. thick layer of Klemocell foam without a backing increased the natural frequency and the damping. The approximate reduction in stress was two-thirds.

3.2.7 Sud Aviation

A test installation consisting of the portion of the Caravelle aircraft aft of the pressure dome has been set up. One engine has been fitted and endurance running is being carried out to check the fatigue life of the structure.

A comprehensive series of noise measurements on the fuselage side of the test rig and the prototype aircraft have been made by O.N.E.R.A. and are reported in detail in Sud-Aviation Reports[69,70]. An extensive series of strain gauge measurements on the fuselage, tailplane, elevator, fin and rudder was also made and statistical analyses of a large number of the strains have been carried out to determine the amplitude distribution. An important feature of the test programme has been the correlation of strain measurements on the test rig (with only one engine running) with strains measured on an actual aircraft. The variation of strain levels throughout the flight plan has also been investigated in detail.

Figure 39 shows the variation in stress during the initial stages of the take-off. It is seen that with the rear engine configuration the stresses do not rise significantly as the aircraft angle to the ground increases at unstick. Measurements for wing-podded configurations on the other hand indicate (as shown by the dotted line) that the stresses in the tail unit increase at unstick due to reflections from the runway. A typical comparison between stresses at take off and during cruise is shown

in Figure 40 from which it can be seen that the stresses during cruise are only about one-third of those at take-off. On a fatigue basis therefore they will be negligible. The stresses during cruise will be a combination of stresses due to jet noise and also stresses due to boundary-layer pressure fluctuations. For an aircraft flying at a higher E.A.S. the boundary-layer components will be increased in direct proportion to the increase in dynamic head 'q'. The variation in stress with engine r.p.m. is shown in Figure 41(a).

In order to check the effect of two engines running on the actual aircraft with the case of only one running on the test rig measurements were made on an aircraft with one and then two engines running. The difference in stress at any one point between these two conditions is shown in Figure 41(b). From this it can be seen that typically there is an increase in stress of the order of 6% for the two-engine case.

Detailed statistical analyses have been carried out to determine the amplitude distribution of the stresses. Plotting the results on a log N vs $(\text{stress})^2$ basis the points fall closely on a straight line indicating that the distribution closely approximates to Gaussian. There is no significant asymmetry in the curves.

The effect of thrust reversers has also been investigated on a production aircraft. As would be expected, the areas of high stress have changed due to the change in pressure pattern but the maximum levels are no higher than those in the normal case. Thus the main effect of the reverser is to extend the area over which significant stresses due to acoustic pressures exist.

Some analyses of the forms of panel vibration taking place is being carried out by Southampton University (Sec.3.2.5).

3.2.8 D.V.L. (Mulheim)

Some noise levels and spectra have been measured in the near field of a 4000 lb thrust jet engine with a view to developing the measurement techniques and acquiring first-hand experience of the problem. Similarly some panel fatigue tests have been run to gain experience of the different types of structure. Panels 2 ft × 6 in. simply-supported along the shorter side were excited at resonance by an electromagnetic vibrator. Steel honeycomb and foam-filled sandwich were used in addition to open sandwich with corrugated webs. The damping ratio in the case of the riveted sandwich was of the order of 5% and in the case of the bonded or brazed sandwich 2%.

A programme of fatigue testing is in progress to compare the effects of random and discrete loading on fatigue life. The first stage of this work has been reported by Kowalewsky[71], but detailed discussion of the results is outside the scope of this report. Current tests are aimed at finding the effect of subsidiary peaks in an applied waveform by using a sine wave plus the third harmonic. By varying the relative amplitude of the two components a composite wave having different sized secondary peaks can be obtained. To date the amplitude of the 3rd harmonic has been increased to a magnitude equal to the magnitude of the fundamental term. Even at this stage the secondary peaks have no measurable effect - it is the maximum amplitude which fixes the life.

Future plans are for r.m.s. S-N curves to be produced using Gaussian distribution of strain (from white noise generator).

Considerable theoretical and experimental work is being done on the stability of thin shells by Thielemann (who is A.G.A.R.D. co-ordinator on shells). As the stability equations are of the same form as the vibration equations this work should form a good basis for the study of normal modes and natural frequencies of vibrations of stiffened shell structures.

3.2.9 Rome University

No work has yet been done in this field but there is considerable interest in the random vibrations of shell structures. Theoretical work on structural vibrations due to very high rates of heating (re-entry conditions) has been done.

3.2.10 N.A.S.A., Langley

Recent work is aimed at comparing the response of panels to discrete-frequency and random noise in the range of levels from 140 to 161 db.

A random noise test installation has been made from a 12 in. diameter cold air jet operating subsonically[72]. The test panels are placed close to the jet in order to be in the high-intensity region but in this position the spectrum peaks at a high frequency. In order to increase the low-frequency component of the spectrum at the test positions close to the nozzle, four 90° bends were introduced into the pipe immediately upstream of the nozzle.

Panels measuring 11 in. × 13 in. were placed 6 in. away from the 15° boundary of the jet in a row of 4 panels. The noise spectrum at the different panel positions varied as shown in Figure 42. This variation probably accounts for some of the scatter of test results.

The panels were of thicknesses in the range 0.02 to 0.064 in. and were mounted rigidly to the support structure. The mounting plate was circular of 1 in. thick aluminium with a cut-out $9\frac{5}{8}$ in. × $11\frac{5}{8}$ in. Attachment consisted of bolts at $2\frac{1}{8}$ in. centres tightened in a fixed sequence to a preselected torque. No attempt was made to provide free-field backing to the panels. They were just left open in the room which was fairly reverberant. In some siren tests the panel mounting was attached to a chamber containing acoustic absorbent material to prevent standing waves and to enable a static pressure differential to be used.

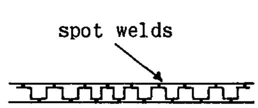
In the discrete-frequency tests only one panel at a time was tested, it being placed 6 in. from the mouth of a 2 ft diameter acoustic horn fitted to the siren. The method of operation was to determine the resonant frequency of the panel and test at that frequency until failure took place. The siren frequency was adjusted as the resonant frequency changed during the test due to crack development. In tests at high intensity and short life the results are less accurate as it took a significant portion of the life to adjust the siren frequency and noise level.

A comparison between random and discrete tests is shown in Figures 43(a) and (b). In the first figure, which shows fatigue life in terms of overall noise level, it is

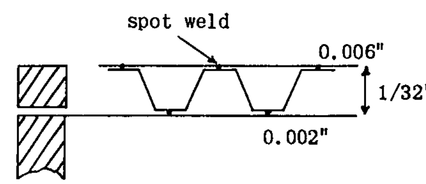
seen that, for a given noise level, life under random excitation is of the order of 10 times as great as life under sinusoidal excitation because in the random case only a part of the energy is available at the panel resonances and hence the efficiency of excitation is less. Where, however, life is plotted against r.m.s. stress level it is seen that the life under single-frequency loading is of the order of 5 times as great as for random loading. Also the fatigue limit for discrete loading is approximately twice as great as that for the random case. This is because in the random stress case the peaks greater than the mean peak do proportionately greater damage than the corresponding peaks below the mean. There is also the additional complication of multi-modal response.

The scatter of results on this basis for the air jet tests is partly due to the different spectra at the different test stations. If the jet results are plotted against noise level in a band of frequencies corresponding to the resonant frequency of the panel then the scatter is reduced. The correlation of the noise pressures over the panels also varies from station to station and attempts were made to measure this. A large panel with transducers let into the surface was set up in the line of the test sections and the noise pressures correlated parallel and perpendicular to the jet axis. However, the pressure cells were only accurate up to 500 c.p.s. and the results were considered to be unreliable.

The work on simple flat and curved panels has now been finished and tests are being prepared for open sandwich type construction of the type being suggested for re-entry capsules. In earlier tests on type (1) (see sketch) the spot welds broke and the skin ripped off. When replaced by rivets an increase of 20 times in life was achieved. A new material about to be tested, North American space metal, again has spot welds, as shown in (2). This is mounted at the edge by crushing the web.



Type (1) spot welds



Type (2) spot weld

In an experiment to determine the noise transmission and stress response of panels to noise a single rectangular plate 0.020 in. thick and 8 in. by 3 in. is mounted on to a small acoustic chamber[73]. The more useful results are obtained when the chamber is lagged with fibre glass to simulate as near as possible free-field conditions on one side of the panel. In the first test configuration the panel is subjected to a high-speed flow of air over its surface from a slit nozzle operating at a Mach number of 0.9. In the second test configuration the panel and supporting chamber are lowered 3 in. from the nozzle and the panel is subjected to essentially near-field jet noise.

The pressure fluctuations measured on the surface of the panel with flush microphones show broad-band spectra in both cases with an overall level of 133 db with flow attached and 125 db for the jet noise case. The spectrum for the flow-attached case

if plotted as spectrum level against frequency would be reasonably flat up to about 2000 c.p.s. before starting to fall off. This is therefore a reasonable representation of a boundary-layer pressure spectrum. In the jet-noise case there is an unrealistic high-energy content in the high-frequency bands up to 10,000 c.p.s.

The strain response of the panel for the two types of excitation shows that in the jet case the fundamental mode is the predominant one to be excited whereas in the boundary-layer case a group of higher modes in the region 1000 to 3000 c.p.s. are also excited. The analysis equipment (1/3 octave) is not good enough to determine how many modes there are in this 1000 - 3000 c.p.s. peak. These modes have presumably not been excited in the jet-noise case because the pressure correlation length would be much greater than the structural mode length.

The noise transmission measurements show that in the jet-noise case the transmission loss approximates to the mass law with slight increases in transmission at the structural resonances. For the flow-attached case, however, the curve begins by following the mass law at low frequencies, has two marked dips corresponding to the two broad panel resonant peaks at 500 and 1000 - 3000 c.p.s. and then continues to fall for higher frequencies. This indicates that the high-frequency noise from say, 5000 c.p.s. upwards, is being transmitted through the panel with much less attenuation than would be expected from mass law considerations. This is likely to be due to the coincidence effect put forward by Ribner and is the first experimental verification of the effect and of the order of frequency at which it becomes important.

3.2.11 W.A.D.C.

3.2.11.1 Aeromedical Laboratory

Von Gierke has developed a random noise siren for use primarily in subjective work on pilots of Mercury project[74]. It could, however, be used for structural work. It has the advantage of being more efficient than the Altec Lansing air modulator but the disadvantage of a fixed spectrum. A 1-inch air jet is interrupted by the teeth of four wheels each rotating at slightly different speeds. The teeth are of unequal width and also spaced unequally. As a result the air nozzle is opened and closed in a random manner as the period over which the whole cycle repeats itself is very long. The frequency range is 200 - 10,000 c.p.s. and an intensity of 165 db in the plane of the exponential horn outlet is achieved. A smaller discrete-frequency siren giving 156 db at the horn exit is also available at this laboratory.

3.2.11.2 Aircraft Laboratory

No panel testing or other similar development work has been done at W.A.D.C. Concern is largely with military aircraft and the policy has been to insist on a 10-hour proof test at full thrust of complete aircraft with all services functioning. In spite of the advances being made in methods of designing structure to withstand satisfactorily acoustic loads it is felt that it is still necessary to have a large-scale test to prove electronic equipment, control systems, etc. When structural failures occur during tests the general procedure is to stiffen the structure and then continue. Damping material is only used as a last resort. No strain measurements are requested although in some cases the contractor himself may make some measurement. A manual of good design practice has been produced for military aircraft and deals mainly with detail design.

A large-scale siren test facility is in the course of construction using an old low-speed wind tunnel building and air supply; consultation and development work is being carried out by B.B.N. The chamber is 70 ft x 50 ft x 42 ft and the aim is to be able to test a large section of an aircraft or complete missiles under intense noise conditions. The noise source is to consist of 12 pairs of sirens mounted in one corner of the chamber each capable of being modulated ± 25 c.p.s. about its selected centre frequency. All 24 could be used in this manner to give an overall frequency range 50 c.p.s. - 10,000 c.p.s. Alternatively all could be operated together at a single discrete frequency.

The room is to be covered with 'random' wedges to make it into a reverberant chamber and there is also to be provision for a fibre glass curtain over all the walls giving 80% absorption to cut down reflections and produce a condition approximating to free field. Unless large quantities of fibre glass are used this approximation is not likely to be very close in the otherwise reverberant chamber. In the reverberant condition it is estimated that a noise level of 160 db throughout will be achieved. In the 'progressive wave' condition it is estimated that 174 db will be achieved over a floor area of 50 sq ft. Both figures are quoted for discrete-frequency operation of the siren. It will also be possible to heat or cool the test specimen from inside.

A small single siren chamber capable of producing 180 db will also be constructed in an adjacent room for tests on electronics and small components.

3.2.11.3 Materials Laboratory

This laboratory has a comprehensive series of fatigue tests in progress. Discussion of these is outside the scope of this report.

3.2.12 B.B.N.

Bolt, Beranek and Newman have a wide interest in the field of structural response to noise and have acted as consultants to many of the aircraft companies. Their own projects at the moment are mainly in connection with the large siren facility being designed for the aircraft laboratory of W.A.D.C. The system is designed to produce up to about 174 db at frequencies from 50 to 10,000 c.p.s. with modulation of any test frequency in a band of 0 to 50 c.p.s. Two sirens are mounted on to manifolding which attaches to the test section ducting. A high-frequency siren operating from 500 to 10,000 c.p.s. is mounted perpendicular to the flow direction and its sound is reflected by an acoustic mirror into the test section. A low-frequency siren operating from 50 to 2,000 c.p.s. produces sound which is transmitted through the mirror to the test section. Either siren can be operated separately to produce a pure tone or together to produce a combination. A prototype is running at Boston with a high-intensity working section to take panels about 2 ft x 1 ft. An exponential expansion downstream leads to a wedge termination. In the expansion electronic equipment can be tested at somewhat lower intensities.

The basic theoretical and experimental work on the damping of plates by means of damping tapes was completed in 1958 (Ref.61). Further developments of damping treatments have led on to 'space damping' where an Araldite honeycomb spacer $\frac{1}{4}$ in. - $\frac{3}{8}$ in. thick is placed between the plate and the foil-backed visco-elastic layer. The effectiveness of this method of increasing the shear in the damping layer has not yet

been evaluated on typical structures. Alternative forms of this system use foam instead of honeycomb as a spacing material. Theoretical and experimental work is also being done on the damping of plates by fibre-glass blankets. A theory has been built up on the study of the wave motion of the air in a very thick blanket excited by flexural waves in the plate. Fair agreement between theory and experiment was obtained in the higher frequency range (800 c.p.s. upwards) but the theory over-estimates the damping in the lower-frequency range. The theory is being extended to consider finite blanket thickness and motion of the blanket itself.

3.2.13 University of Minnesota

A study of material damping has been in progress for several years and basic work on the energy required to fatigue simple specimens has been completed by Lazan and his co-workers. The damping work has recently been extended to cover items of specific application to the acoustic fatigue problem[75].

Theoretical and experimental work is being carried out to determine the energy dissipation produced by a visco-elastic interface at joints. In the idealised case considered a built-in beam was mounted with a visco-elastic interface at its support. The theory is being extended to plates.

Another project is considering the energy dissipated by a damping layer (of the tape form) spaced from the test beam by foam in the form of a corrugation.

Lambert and his co-workers are making a theoretical analysis of the response of a uniform bar to broad-band noise. The theoretical work is complete and experimental verification on a built-in beam is proceeding. To date two point sources have been used at ¼ and ¾ span positions with a continuously variable time delay between them. Damping tape was added to the beam to investigate the variation in response with increase in damping.

3.2.14 U.C.L.A.

Powell continues to work in consultation with Douglas on theoretical aspects of noise production and structural response. Most of his noise work relates to far-field spectra and intensities.

Theoretical work is aimed at the development of the normal mode method of calculating structural response as originally set out in References 76 and 77. Present work is studying the correlation between modes, the influence of pressure correlation patterns, and the application to complex structures.

Leonard (Physics Dept.) and Rudnick are authorities on sirens and have acted as consultants on the designs of several of the high-intensity discrete-frequency sirens in operation with the aircraft companies. For maximum efficiency the nozzles should be designed to give subsonic velocities of the order of 400 ft/sec. There is also an optimum shape for the apertures, to produce a sinusoidal pressure wave. 30% efficiency is achievable and 2 k.w. acoustic power relatively easily attainable.

3.2.15 Avro Canada

Initial work began with the need to provide a 'fix' for the C.F.100 aircraft and was later extended to design development for the structure of the C.F.105 'Arrow' supersonic fighter. A siren was built to provide the main noise test facility and some 200 panels tested in the first phase of the work. The later work developed into a research programme but was discontinued in February 1959 before it was completed.

A simple air chopper in which the teeth of a wheel interrupted the air from a single jet formed the noise source for the panel tests. The jet pressure had a maximum value of 100 lb/in.^{2} (factory air supply) with a simple reducing valve to obtain lower noise levels. An exponential horn 6 ft long of square cross section and having an exit size of 1 ft x 1 ft was used to produce a plane wave. A test stand 2 ft x 2 ft was placed 6 - 10 in. away from the horn at normal incidence to the sound wave giving noise levels on the panel of the order of 170 db.

Initially panels 6 in. x 3 in., and some 11 in. x 6 in., were tested rigidly mounted in the centre of the 2 ft x 2 ft frame. The aim was to gain experience of the type of failure possible and assess the value of different types of material. As in other cases reported, testing was carried out at resonance. No attempt was made to convert test lives to aircraft lives at the lower noise level of 140-150 db.

An interesting series of tests was carried out on a larger section of the lower fin. The structure consisted of 3 rigid parallel double web spars with light (0.040) ribs and 0.051 skin. The spar and rib spacings were 12 in. and 8 in. respectively. The test panel consisted of one skin, 2 spars and 3 ribs, with a second dummy skin to stabilize the spars and ribs. This dummy skin was cut away in the centre of panels to allow inspection of the inner skin of the panels being subjected to noise. It was found that no free edges could be tolerated and spar and rib webs were bolted rigidly to the stand. The structure was then tested at its main panel resonant frequency at 170 db. Typical service failures were reproduced on this rig and can be listed:

1. Rivet failures

2. Rib flanges cracking - improved by symmetrical attachments

3. Discontinuities - rib flange joggles at spars.

Reduction of panel size by the addition of light Z stringers parallel to the spar was ineffective and heavier stringers gave trouble at stringer/rib intersection.

The final solution was to reduce the rib spacing from 8 to 6½ in. and *reduce* the skin thickness from .051 to 0.040 in. with a doubler 4 in. wide (2 in. each side of rivet line) cold pressure bonded at 700 lb/in.^{2}, with ordinary bonding elements 0.002 in. thick, to the skin. This provided local stiffening in the region of high bending moments and stress concentrations and also increased the damping of the structure. The width of doubler was found to be critical, larger or smaller dimensions giving rise to failures at the end of the doubler. The maximum amplitude of the thinner skin was *reduced* from 0.3 in. to 0.15 in. at 170 db and the life increased from 15 - 20 minutes to 3 hours. It was concluded from these results that tests on single panels rigidly fixed along their edges were ineffective in reproducing practical failures and that larger sections must be tested.

The increase in test level from 140 db to 170 db gives rise to many problems - the effect of non-linearities perhaps being the greatest. At the aircraft level of 140 db the panel would only just be in the non-linear range but at 170 db with 0.15 in. deflection of 0.04 in. thick skin considerable non-linearities would be present.

As a result of this work a research programme was initiated by L.H. Gould. The initial objectives were to estimate the response of a simple panel of infinite length by a consideration of the energy balance of the panel. A theory has been developed and reported in Reference 78. Tests to study mode shapes, stress distribution and stress-time variations in the non-linear range were made on panels 6 in. long and 3 in. wide clamped on the short edges, and excited by the siren. The effect of added damping materials was studied in a series of tests on cantilevers 6 in. x 0.7 in. x 0.093 in. tested by an electro-magnetic vibrator. Various combinations of doublers and damping layers near the root were tried out. This work was discontinued in February 1959.

3.2.16 Boeing

Extensive test facilities include a discrete-frequency siren on which much of the early panel work was done. This is a normal incidence configuration giving pressures up to 182 db over an area of approximately 2 ft x 2 ft. Panels are tested at the first resonance but in some cases this resonance may be an air resonance of the horn. After the early ad hoc work more controlled experiments are being carried out. An 'anechoic' box is placed behind the test specimen giving an inside level 20 db below the outside level in the range 100 - 500 c.p.s. This reduction is of the same order as that which occurs on aircraft structures. The installation is situated in a large chamber isolated from the surrounding buildings by a vacuum space. This is used for comparative testing of different materials and no direct attempt is made to calculate an aircraft service life.

Lin has carried out a theoretical analysis of air column resonance phenomena in normal incidence testing[79] using a theoretical model of a vibrating infinite rigid plate to represent the sound source. The result of this theoretical investigation is a proposed test technique which will eliminate or greatly reduce the errors due to horn resonances. The basis of the recommendation is that a calibration test be carried out initially in which a rigid panel is put in the position to be occupied later by the test panel. The siren is then calibrated in terms of blowing pressure to give a constant excitation pressure on the rigid panel with frequency. When an air column resonance is reached the required blowing pressure to achieve a given noise level will be much less than that required at frequencies away from resonance. In the actual test with a flexible panel the excitation pressure will be deduced from the blowing pressure rather than from a microphone in front of the panel. Whilst this procedure overcomes the gross errors associated with uncorrected normal incidence tests the accuracy to which the excitation pressure is known is likely to be less than in the case of grazing incidence tests where a microphone can be placed near to the specimen.

A more extensive and realistic test facility consists of a reverberant room adjoining an anechoic chamber. A 'window' 6 ft x 6 ft is left in the common wall and large pieces of structure can be mounted in this space. The mode of operation is to use a random noise siren in the reverberant chamber to produce high-intensity random

noise on one surface of the test section. This is used for transmission loss measurements and also structural response tests although the levels are not high enough for some of the fatigue testing. The random noise siren (Altec Lansing) produces 150 db in the reverberant room and is flat over the four octaves 100 - 800 c.p.s. The input spectrum can be shaped to produce any noise spectrum in that frequency range. The reverberant room is also used for environmental testing of electronic components in a diffuse field - reasonably representative of the conditions inside an instrument compartment.

An engine test cell has been used for the assessment of the wing trailing edge structure of military aircraft. A large section of the wing is mounted beneath the twin-jet pod test installation. The trailing edge structure is honeycomb sandwich with the edges squashed to make the attachments. Initially failures occurred at the squashed edges - this was cured by the injection of adhesive along the edges.

Theoretical work is being done by Lin[80] of the Structural Dynamics group on the normal modes and response of stiffened structures. Experience of failures on the fuselage of aeroplanes having conventional Z stringer-frame construction indicated that the predominant type of vibration was one in which panels and stringers vibrated between frames. Thus on the basic assumption that the frames form a boundary to the vibration Lin developed a theory to give the normal modes and frequencies of a skin/stringer panel simply supported on two frames. The first type of vibration is with stringers twisting and adjacent panels vibrating out of phase. In the second type the stringers bend and adjacent panels also deflect but are in phase. A combination of higher panel modes with either stringer twisting or bending gives a series of modes and frequencies for the structure. The theory was applied to a typical fuselage structure and a response test was carried out in the reverberant/anechoic room. The comparison between measured and theoretical results is shown in Table II.

The agreement here is seen to be very good, but for slightly different types of structure this is not quite the case as can be seen from the Southampton results on the Caravelle. However, it forms a very useful basis for a reasonable mathematical approach and experiments on different types of structure to assess its general application should be made.

Using these simplified mode shapes for the skin vibration Lin has gone on to develop a method for estimating the stresses induced in the skin by jet or rocket noise[81]. The method is based on Powell's normal mode approach and gives an order of magnitude agreement for r.m.s. stress although the computed stress spectrum is in error for the higher band of frequencies. The modes of vibration of the skin/stiffener panels have closely spaced frequencies and therefore the response in each mode cannot be considered to be independent of all other modes. Lin puts forward a simplified way of dealing with this cross coupling for a lightly damped system. For a forcing frequency slightly lower than the lower limit of a frequency band all the modes in the band respond almost in phase with the excitation and the amplification effects of all the modes are additive, giving a maximum power spectral density near such a frequency. For a forcing frequency slightly higher than the upper limit of the band all the modes in the band respond out of phase with excitation, again giving a maximum response. When, however, the forcing frequency lies in the centre of the band half the modes will respond in phase with the excitation and half out of phase and the total response will be a minimum. Even this approach involves a prohibitive

amount of computation and Lin simplifies the method still further by computing the response of only a few of the modes and then averaging them out to represent the average contribution from each mode in the band. Using this approach the results obtained are as shown in Table IV. Typical pressure correlation lengths were taken from the work of Howes[10].

Recent theoretical work is aimed at finding the effects of skin thickness, differential cabin pressure, and structural damping on transmission of noise into cabins from turbulent boundary layers. A preliminary study has determined the effect of these parameters on the transmission loss of a simply-supported panel subjected to grazing incidence discrete-frequency acoustic excitation. The differential cabin pressure has been shown to be a very important parameter.

The company has also concerned itself with fatigue under random loading and recent work by Fuller[82] has a bearing on the problem of the estimation of fatigue life of structures subjected to random acoustic loading. A detailed discussion of this work is outside the scope of this paper.

3.2.17 Convair

An extensive series of siren tests has been carried out and Getline claims to be able to predict life on an aircraft from specimen tests. Honeycomb sandwich structure is used extensively in regions of potential noise damage. At attachment points the honeycomb is milled out and extruded sections inserted. To prevent moisture freezing in the cores and causing failures the matrix is sealed off - pressure differential is not significant.

The test facility at San Diego is essentially a discrete-frequency siren, designed in consultation with Leonard (U.C.L.A.), with an exponential horn leading to a working section of the grazing incidence type. Test specimens up to 4 ft x 2 ft can be accommodated and the maximum level attainable is 171 db with a mean air flow of 30 miles/hr in the duct. The sound field in the duct has been investigated and, with a rigid boundary surface installed in the test area, shows a variation of about ± 1 db over the entire working section at a working pressure of 165 db. At this working pressure, which is essentially sinusoidal, the second harmonic is down at least 10 db below the fundamental. Specimens being currently tested consisted of panels with typical frames dividing the section into 1 ft wide bays. The frames have a thermoplastic interface between the flange and skin mainly for sealing purposes. The edge supports were made of tufnel to enable standardised mounting conditions to be repeated. Rivet failures were being studied with different types being tested. Nonlinearities of response were less marked in the sandwich panels than in sheet metal specimens.

Getline's method of prediction of aircraft fatigue life from siren tests[83] is based on the following assumptions:

(1) Response in *uni*-modal and the damping in this mode is small;

(2) Response therefore is of form of a discrete-frequency corresponding to the resonant fre~uency of the panel with an amplitude varying randomly;

(3) R.m.s. level of amplitude can be determined from input pressure (Miles relationship);

(4) An S-N curve for the structure is available and normalised on a stress ratio basis;

(5) The pressure input is defined in the 1/3 octave around the resonant frequency on a basis of the probability of the occurrence of peak values.

The test panel representative of, say, wing underskin is mounted on the siren box in a way 'as nearly representative as possible of the aircraft mounting'. In practice this means fully fixed round all edges as this is the only condition which can be reasonably consistently reproduced. Trailing edge sections (e.g. flap or aileron) may be installed with the downstream section unsupported, and merely contained in the siren by a pressure seal. A siren frequency sweep is then carried out at a constant but reduced amplitude to determine the mode in which the highest response occurs.

Having determined in this way the test frequency the next step is to decide on the noise pressure level which will represent the aircraft conditions. The procedure adopted is open to considerable doubt but seems to be justified by the fact that 'it works'. The basic assumption is made that the statistical distribution of peak stresses can be obtained directly from the applied pressure amplitude distribution. This is true for Gaussian distributions but has not yet been established as being true for any arbitrary pressure amplitude distribution. However, on this assumption a plot of essentially peak stress vs number of applications is made. Then comparing this curve with the S-N curve it is possible to determine which stress level (and by inference which pressure) causes the highest proportion of damage. The test is then run at this pressure level (care is taken to ensure that the pressure is not so high as to cause non-linear strain response). When failure occurs the interpretation of the result is that if the test pressure occurs only 2% of the time, say, then the life under service loads (100% time) will be 50 times as great as the test life. This assumes that the lower pressures do not produce any significant damage.

Aircraft in airline service have now had the equivalent of 30 hours at full thrust (1500 hours flight time) without a single incident of acoustically induced fatigue.

At Fort Worth full scale proof tests of the type described in Section 3.2.11 are being carried out for Military certification purposes.

3.2.18 Douglas

Proof Testing has been carried out in detail on the RB.66 aircraft. An accelerated test on the whole aircraft by testing at low ambient temperatures was conducted in Alaska. Noise levels were increased by 3 db on average, thus making the 10-hr test equivalent to 150 hours at standard operating conditions. In this test opportunity was taken to make comprehensive strain gauge measurements on the structure[84,31]. Power spectra of the noise pressures and the structural response[84] show that in many cases the strain response is multi-modal in form. General conclusions were that in the fuselage case of widely spaced stiff frames (24 in.) and relatively thin skin, the response was essentially uni-modal in form. However, in the case of the flaps, elevator and rudder, where the rib spacing is of the order of 4 in. and the skin is

relatively thick, the response as measured by strain gauges on the skin showed a multi-modal form.

The comprehensive series of measurements of noise spectra show that although the overall level is increased by 3 db, at low ambient temperatures the spectra at given positions is changed somewhat from warm air conditions. Measurements on the rudder skin show that with only one engine running the sound pressure on the shielded side is 18 db less than that on the exposed side but the skin stress on the shielded side is only reduced by 1½ db (1200 - 720 lb/in.^{2}). With both engines running the increase in stress over single-engine running is equivalent to an increase of about 3 db in a single source.

Engine cell tests were also carried out on rudders and elevators mounted in a noise field 4 db above the standard level. This was considered to be equivalent to the 10-hr test at Alaska with 1 db added to cover additional uncertainties in the noise field around the engine. The basis of comparison was the r.m.s. skin stress on the aircraft and on the cell rig.

Civil Aircraft (D.C.8 structure)

This represents one of the first attempts to design ab initio structure to withstand acoustic loads[85,86]. A theory has been developed from Miles's work[87] on single-mode response to enable life under random conditions to be computed from high-intensity siren tests. The essential steps in the procedure adopted are:

(1) Acoustic loads are determined from extrapolation of near field data on engines of similar size.

(2) Computation of stresses in a single mode caused by jet noise from data obtained using a siren (assuming linear response):

$$\sqrt{(\overline{S^2})} = S_H \frac{P_R}{P_H} (\pi \delta f)^{1/2} \tag{1}$$

(3) Modification of (1) to allow for non-linearities by the factor λ (see Fig. 44(a)).

(4) Further modification of (1) to allow for multi-modal response by the introduction of the factor γ :

$$\gamma = \frac{\sqrt{(\overline{S^2})}}{\sqrt{(\overline{S_1^2})}}$$

(5) The relation between siren stress and jet-noise stress is now

$$\frac{\sqrt{(\overline{S^2})}}{S_{H_1}} = \lambda \frac{P_{R_1}}{P_{H_1}} (\pi \delta_1 f_1)^{1/2} \gamma_1 \tag{2}$$

(6) Miner's Cumulative Damage hypothesis is assumed to be valid.

(7) If the amplitude distribution is given by the Rayleigh probability distribution then the number of cycles to failure under random load is:

$$N_R = \left[\int_0^\infty \frac{P(x)dx}{N_x}\right]^{-1}$$

(8) Then

$$\frac{\sqrt{(\overline{S^2})}}{S_{H_1}} = \frac{\phi(N_R)}{\phi(N_H)} \qquad (3)$$

The appropriate S-N curve together with Equations (2) and (3) is used to determine the allowable S.P.L. for the structure loaded by jet noise in terms of data from siren tests.

In the above,

f	=	frequency, c.p.s.
N_R	=	number of cycles to failure under random load
N_X	=	number of cycles to failure under cycle loading at level
P_H	=	r.m.s. sound pressure for discrete loading
P_R	=	r.m.s. sound pressure in a one c.p.s. bandwidth
$P(x)$	=	Rayleigh probability density function
$\sqrt{(\overline{S^2})}$	=	r.m.s. stress due to random loading
S_H	=	r.m.s. stress at resonance for load P_H
γ	=	multiple-mode factor
δ	=	damping ratio
λ	=	non-linearity factor
$\phi(N_H)$	=	allowable stress at constant amplitude for a given life N_H
$\phi(N_R)$	=	allowable stress due to random load for a given life N_R

Suffix i indicates the fundamental or predominant mode.

A Nomograph has been constructed to show this (Fig.44(b)) and also a tentative design chart has been produced to give desirable rib pitches and skin thicknesses for any noise level.

Although there are many criticisms of this form of analysis it has enabled structure to be developed which has withstood full-scale proof testing on a jet rig. The siren section was large enough to enable 2 ft x 2 ft sections of structure to be tested at grazing incidence. After initial tests on plane panels alone, sections having typical ribs and stiffeners were used and development of detail attachments was possible.

The aim was to achieve a fatigue life of 1000 hours at take-off power, representing 50,000 hours flight time.

As would be expected, reasonably close agreement between estimates and practice was achieved in single-mode cases but the agreement decreased as the number of modes increased.

3.2.19 Lockheed

In parallel with studies of the acoustic pressure field close to a jet engine Lockheed have initiated a programme of structural response investigations. Structural panels of skin/stringer construction were mounted near the exhaust and strain measurements made. At the time of the author's visit experimental results were not yet available.

3.2.20 Minnesota Mining and Manufacturing Company

The Company manufactures the basic visco-elastic material used in shear damping configurations. The layer is available on different thicknesses of foil backing. It gives maximum energy dissipation at room temperatures and has limits of operation at -20°F and $+150^{\circ}$F. The company carry out basic tests on the material to determine the dynamic shear modulus under different temperature conditions.

3.2.21 North American

The Company is in the process of setting up a large discrete-frequency siren facility of the grazing incidence type. No structural work has been done to date although Belcher has joined the company and proposes to use similar methods of analysis to those he developed at Douglas (Sec.3.2.18). The siren facility consists of two large sirens with the possibility of modulating them to give a band of frequencies. Close to the sirens is a test section to take 3 ft x 5 ft specimens with 170 db in the frequency range 50 - 10,000 c.p.s. Further downstream is a horn into which a larger piece of structure can be placed with an anechoic backing box. The level on this section is to be 162 db and its angle can be changed relative to the sound waves to investigate the effect of sound trace wavelength.

Considerable test work has been done at Columbus, Ohio, on the effect of lower-intensity noise on electronic components, where the facilities include a large array of loudspeakers giving a noise level of the order of 140 - 145 db dependent on frequency. A reverberant chamber using a cold air jet as a noise source and having dimensions 5½ ft x 5½ ft x 6 ft is being developed to give 155 db for instrument testing. A large siren being developed at Columbus, again of the grazing incidence and discrete-frequency type, has a test section 14 ft x 4 ft x 1 ft, although normally specimens will be of the order of 4 ft x 3 ft. The level on this size of specimen will be 170 db.

Fuller details of this wide range of test facilities is given in the Proceedings of the Environmental Engineers[88].

3.3 DISCUSSION

This review reveals that, although the major aircraft companies and research establishments have been actively concerned with acoustic fatigue for anything up to 7 years, no generally agreed approach to the solution of the problem has been reached. There is still a considerable variation in experimental and analytical methods and finally recourse is often taken to full-scale tests of a complete aircraft or at least a substantial portion of structure and systems in the vicinity of the efflux. The theories of structural response to random pressure loads have not been sufficiently far advanced to apply to practical structures and hence several semi-empirical approaches have arisen to cope with the immediate practical situation. As a result of these different philosophies different testing procedures have developed and varying success is claimed by the respective authors. In certain aspects of structural response, therefore, it is possible to quote generally agreed results whereas in many other cases one can only catalogue and comment on the different methods with the hope that more directed effort will be brought to bear on the problem.

3.3.1 Theories of Structural Response

The theoretical approaches which have been used can be classified into two categories:

(1) Equilibrium of forces

(2) Energy balance.

In the first case the sound pressures are related to stresses in the material and the stresses in turn to fatigue life. In the second case an attempt is made to relate the energy of the impinging sound to the energy required to cause fatigue failure of the structure.

The majority of work to date has been concentrated on the force method, which is further subdivided into

(a) Normal mode approach

(b) Wave approach

The basic relationships for the normal mode approach to the problem were set out by Powell[57,58] for a complete structure and by Miles[87] for a single degree of freedom linear system and have later been developed and extended by others for special cases. It is true to say that an apparently workable method has been formulated in terms of the normal modes of the structure but that the practical difficulties arise from the lack of knowledge of the normal modes and damping of typical structures. At first sight it would seem that the large number of possible modes of vibration of the structure would each have to be considered, but in practice it has been found that the major response has been limited to a few modes (up to about ten in some cases).

Miles's uni-modal theory has been used in earlier tests by the N.A.C.A. Langley workers and good agreement has been obtained between theoretical and experimental stresses for single panels excited in the fundamental mode of vibration. In the more recent experimental work using the air jet a multi-modal response occurred and consequently Miles's work could not be applied.

In the aircraft industry in the U.S.A. Miles's theory has formed the basis of several semi-empirical methods used to predict the fatigue life, of a structure exposed to random acoustic excitation, from fatigue data obtained under siren (discrete) excitation. The Rayleigh distribution of stress peaks is used together with Milner's cumulative damage hypothesis to relate life under random conditions to sinusoidal conditions on the siren test.

Belcher and his co-workers (Sec.3.2.18) modify the single degree of freedom linear analysis by two factors. The first is to allow for non-linearities likely to be present in the siren test to a greater extent than in the aircraft case, and the second is to allow for the stress increase due to responses in higher modes than the fundamental. The values of these two factors cannot be obtained in general terms and must really be regarded as empirical. A reasonable estimate of the effect of non-linearities appears to be feasible but the multi-modal factor is indeterminate. This is confirmed by the Douglas experience that the degree of agreement between estimates from siren tests and actual aircraft lives decreases as the number of modes, having appreciable response, increases. As more experience of the types of vibration taking place in typical structures is built up it may well be possible to define this second factor more precisely. The effect of fully fixing the test panels has not been included, and therefore on single panel tests an additional factor to allow for mounting conditions seems to be necessary.

An alternative method sponsored by Getline (Sec.3.2.17) is less clear in detail but appears to be based on the doubtful assumption that the strain peak distribution is directly and simply related to the pressure peak distribution. It further appears to be assumed that the most damaging stress is the predominant one in determining the life and that lower stresses make negligible contributions to the fatigue life. No allowance is made for multi-modal response. The method, however, is claimed to give satisfactory results on the Convair 880 aircraft.

An alternative method using the wave approach is being explored at Southampton University but it is too early to say whether it will be possible to extend it to apply to typical structures. It appears likely that it will be just as difficult to apply to stiffened structures as the normal mode method.

Another alternative method put forward by Gould[78] and also by Regier and Hubbard[89] is based on the energy balance in the structure. As the method has not been fully investigated yet it is not possible to assess how effective it will be in estimating fatigue lives. Much of the information required for the normal mode method will also be required for this method to enable energy dissipation at the joints and by acoustic radiation from a typical stiffened structure to be calculated. It looks sufficiently promising, however, to justify further development.

3.3.2 Modes of Vibration

Compared with the effort put into other aspects of acoustic fatigue little work has been done on the theoretical or experimental study of the modes of vibration of stiffened cylinders. Early work on cylinders with uniform wall thickness has been extended to the case of stiffened cylinders whose stiffening is closely spaced relative to the structural wavelengths. No work, however, has been completed on the case of stiffener spacing of the same order as the structural wavelength, although such a study has recently been started at Southampton University.

An experimental study of the modes of vibration being excited by jet noise has been started by Southampton University in conjunction with Sud Aviation and R.A.E. The first tests on the Caravelle rear fuselage have shown that a relatively simple form of vibration in which the frames act as boundaries appears to be taking place. This is being further investigated. However, this applies only to one particular form of construction and cannot be expected to apply generally. As this information is of paramount importance in assessing the validity of any smaller scale test procedure and determining the optimum position for any added damping layer more such tests should be carried out on different types of structure.

A theoretical analysis of the modes of vibration of a sheet/stringer combination vibrating between two frames as boundaries has been made by Lin (Sec.3.2.16). He obtained very good agreement with experimental results on a section of the Boeing 707 fuselage. When this method is applied to the Caravelle results the agreement is not quite so close, as can be seen from Table IV.

The Energy method has been used by Ford at Southampton University (Sec.3.2.5) to analyse this type of vibration and as can be seen from the table there is close agreement with Lin's theory. Lin's original work omits the stringer energy but when this is included the main effect is to raise the fundamental frequency by about 15%. The main advantage of the energy method is that it can readily be used to allow for varying combinations of 'twisting' and 'bending' stringers and in this way a range of frequencies is seen to exist between 727 c.p.s. and 1086 c.p.s. for the Caravelle Panel 5. The strain spectrum for Panel 5 indicates that such a range exists in practice.

3.3.3 Damping

Since it was recognised that the structural response to random excitation would be resonant a considerable amount of work has been done (Secs.3.2.4, 3.2.5, 3.2.6, 3.2.12, 3.2.13, 3.2.20) on the study of damping mechanisms present in an aircraft structure and methods of increasing the damping. Some measurements of the overall damping in a structure from all sources are given in Table V. In some cases these will be overestimates due to the bandwidth of the equipment.

These results indicate that the overall damping ratio in an untreated conventional structure is likely to be of the order 0.015 - 0.025. This is made up essentially of three components:

(1) Structural hysteresis in the basic metal itself

(2) Structural hysteresis at the numerous joints

(3) Acoustic radiation.

Lazan's extensive study of item (1) (Sec.3.2.13) indicates that its contribution to the overall damping ratio is very small for the case of single aluminium skins. Preliminary measurements of the damping of honeycomb skin sections (Sec.3.2.5) show that the basic structural damping is some two to three times as great as for a single skin. Even these values, however, are still small and make little contribution to overall factors of the order of 0.02.

The relative magnitude of items (2) and (3) is not known at the moment. Earlier work by Mead suggested that the damping ratio due to joints around the edge of a single panel is of the order of 0.003. This damping varies with rivet load and therefore with mode shape. An additional source of this type of damping is the attachment of the skin to the frames or ribs but it does not seem likely that this will appreciably increase the damping factor. At most a factor of the order of 0.005 in panel type modes seems to be predicted.

This implies that a large part of the damping is due to acoustic radiation. Early estimates, based on Junger's[90] study of the acoustic radiation due to overall modes of vibration of a cylinder, gave values in the range 0.02 - 0.03. This, however, is for structural wavelengths in the range 3 - 10 ft longitudinally and 4 - 15 ft circumferentially, thus being considerably larger than typical panel sizes. At shorter structural wavelengths of the order of typical panel dimensions the acoustic radiation on Junger's theory would be very small. The cut-off point occurs when the structural wavelength is equal to the radiated acoustic wavelength. In conditions of shorter structural wavelengths no radiation takes place. In the case of the Caravelle panels the longitudinal structural wavelength is of the order of 16 in. and the radiated wavelength is of the order of 20 in. - giving zero acoustic damping on the overall theory. Theoretical and experimental work by Mead and Mangiarotty (Sec.3.2.5) on single panels indicates that the acoustic damping is low - of the order of 0.002. This is, therefore, consistent with Junger's theory for normal panel sizes. Theoretical investigations of the damping of a group of panels built up from the single panel theory is being currently carried out to determine whether there are any configurations which do result in high acoustic damping.

Clearly further work is necessary to indicate whether acoustic damping forms the largest part of the basic damping in a structure or whether it is insignificant. It may well be that only in cases where optimum wavelength conditions are reached the acoustic damping becomes important.

A considerable amount of work has been done on the effect of added damping layers. There are two basic types of compound.

(1) Flexible compound used in shear (foil-backed layer of joint interface);

(2) More rigid mica-filled compound such as Aquaplas used in direct stress (applied directly to regions of high direct stress).

In both cases it is necessary to know the mode shapes on the actual structure if the material is to be used in the most efficient way. Considerable variations in procedure exist in the U.S.A., where in one case of a backed layer at least three

different methods of application are used on very similar structures. It is important that the damping tape, dissipating energy in shear, is used under conditions of optimum structural wavelength, otherwise the damping increment may be small. Tapes can give damping factors as large as 0.10 with the thicker tapes. Under more realistic conditions when a panel is covered with a tape such that the added weight is of the order of half the weight of the plate an increment in damping ratio of the order of 0.03 is likely to be achieved. For similar weights of material, Aquaplas used on a plate gives a slightly higher damping increment. Mead has pointed out that a uniform thickness treatment on panels may not be the most efficient way of applying the materials. If the mode shapes are known in detail the materials can be distributed at a point of either maximum shear or maximum direct stress and higher damping ratios achieved.

It appears that basic information on the behaviour of both types of material on laboratory test specimens is now available and being extended rapidly. What is not so clearly known is how the materials will work on a built-up structure. As soon as more information is available the effectiveness of the added materials can be estimated reasonably accurately.

3.3.4 Stress Levels and Spectra

The stress amplitude distribution has been checked by several workers. In particular Sud Aviation have shown that it approximates closely to a Gaussian distribution. The strain spectrum in this case shows a predominant mode and therefore it would be reasonable to assume a Rayleigh distribution of the peak stresses. R.A.E. in unpublished work have also checked the distribution for both uni-modal and multi-modal responses and again shown good agreement with a Gaussian distribution. These checks however have only been conducted for low r.m.s. stress cases (up to 5,000 lb/in.^{2}) For higher r.m.s. stress conditions non-linearities will presumably tend to truncate the high stress end of the distribution curve. However, in cases where a long life has to be achieved the stresses will be relatively low and hence it will be reasonable to assume a Gaussian distribution. The peak distribution used for fatigue estimates is generally assumed to be a Rayleigh distribution. This will be adequate for essentially uni-modal responses but may not be correct for cases of multi-modal response.

Measurements of the strain spectra on aircraft structure close to a jet efflux have shown that in some cases response is almost entirely confined to a single mode whereas in other cases four or five modes may have appreciable response. The overall levels on conventional aircraft have generally been of the order of up to 2000 lb/in.^{2} excluding stress concentrations. A few experimental results are listed in the Table VI and Figure 45 shows some typical RB-66 spectra.

It is not possible to draw any precise general conclusions from such measurements as have been made. It does appear, however, that in many cases of widely spaced frames or ribs (12 in. - 24 in.) or relatively thin skin on stiff frames (Caravelle) the skin strain response may be predominantly uni-modal. In other cases of closely spaced ribs the skin strain response will be multi-modal in form. The majority of measurements of stringer, frame and rib flange strains have shown a multi-modal response. The frequency range of major structural response appears to lie between 100 and 1000 c.p.s. There are no records of any significant response above 1000 c.p.s. known to the author.

3.3.5 Strain Attenuation due to Forward Speed, Suppressors, etc.

Only a small amount of empirical information is available on the effects of engine conditions. This can be summarised as follows:

(1) *Effect of Forward Speed*

Caravelle: More than 10 db reduction in strain levels at the end of the ground run during take-off.

(2) *Effect of Noise Suppressors*

Comet: Rolls-Royce corrugated nozzle gave 20% reduction in maximum tailplane skin strains.

(3) *Effect of Thrust Reverser*

Caravelle: Considerable increase in skin strains in some areas but these areas do not coincide with areas of maximum strain during take-off.

Much more empirical information is required before any general conclusions can be reached.

3.3.6 Test Facilities

By far the largest amount of effort on acoustic fatigue has been directed towards the development of installations and procedures to test critical parts of the structure in a simulated noise environment. A comprehensive review of American facilities has been made by Forney[91]. The value of any particular test configuration must be assessed in the light of the effectiveness with which it reproduces the following parameters:

(1) The noise environment -

(a) Intensity

(b) Spectrum

(c) Correlation

(2) The structural configuration -

(a) Mounting

(b) Damping

(c) Backing condition (sound absorbent or reverberant).

Many full-scale proof tests of complete aircraft have been made in truly representative conditions but this is a very expensive and time consuming process and generally much simpler types of tests have been used in the design development phases of an aircraft

project. The present policy of many of the aircraft companies is to use a small-scale test facility of the siren type for design development and then finally check larger portions of structure behind a jet engine.

The noise environment can be simulated in one of several ways such as:

(a) Jet engine on open-air stand

(b) Jet engine in a test cell

(c) Loudspeaker arrays (for lower-intensity environment for internally stowed equipment etc.)

(d) Air jet

(e) Siren.

In the case of a jet engine on an open-air stand it is not always possible to use the actual engine designed for the aircraft but generally another similar engine is used. The specimen is then positioned closer to the exhaust to get the higher intensities required for the new design. It will not be possible to reproduce the spectra exactly in this way but it should be possible to reproduce reasonably accurately the spectrum in the range from say 150 c.p.s. to 1000 c.p.s., the important structural region. The significance of the change of correlation pattern will depend on the types of modes being excited. It is reasonable to assume therefore that test installations of the type established at the R.A.E. form a valid method of reproducing the noise environment.

Jet engine test cells have been considered in England and used to some extent in America. The difficulty here is that the noise spectrum is usually very distorted due to the reverberant conditions which exist in most test cells. There is also the additional difficulty that in some designs the low-frequency sources downstream in the efflux may be enclosed in the exhaust ducting. In some cases attempts have been made to reduce standing waves by attaching sound absorbent material to the walls. Companies using this method generally strain gauge the structure on the aircraft if possible in order to measure actual operating strains. The specimen is then positioned in the test cell, usually with a sound absorbent backing box, at a position which gives approximately the same r.m.s. strain level. It will however in general only be possible to reproduce the r.m.s. level whereas the strain spectrum will have changed and may thus affect the fatigue life.

Loudspeaker arrays have been used by some companies, notably North American, Columbus, to reproduce lower-intensity environments for tests on electronic equipment, etc. Up to the economic limit of intensity of the order of 145 db a reasonably accurate spectrum can be produced. The Columbus installation produces a wave of sound which passes over the test item from an exponential horn. This will not reproduce exactly the pressure correlation patterns present in an instrument compartment which is likely to be a reverberant chamber.

Where a large compressed air supply is available an air jet is a possible source of high-intensity noise. At NASA, Langley, a 12 in. diameter air jet is capable of pro-

ducing an overall noise intensity of 157 db over a test area of several feet close to the nozzle. The four 90° bends in the pipe just upstream of the nozzle increase the low-frequency content of the jet noise. However, in this installation the noise spectrum changes appreciably in the region close to the jet. A constant spectrum cannot be produced over an area much greater than about one square foot. Larger specimens cannot therefore be tested in this installation. It can however be used for comparative panel tests as described in Section 3.2.10.

The most commonly used high-intensity noise device is the siren. Many companies operate a siren facility as summarised in Table VII. It can be seen from the table that there are considerable differences in design and mode of operation and the use varies from mere comparative testing of panels, as in the early Boeing work, to a complete interpretation of siren life in terms of aircraft operating life (Convair and Douglas).

With the exception of the two lower-intensity random sirens all the sirens are of the discrete-frequency type. This immediately raises the serious problem of how the random noise pressures can be represented by a discrete frequency. Strictly a discrete-frequency forcing of the structure at its fundamental mode of vibration is only reasonable if the structure is only responding to a single mode in the noise environment. If this condition is true, as it may be for some large panels, then the appropriate discrete-frequency amplitude can be selected by methods developed from Miles's single-degree-of-freedom analysis by Belcher and others. Even here the representation cannot be exact, for the uni-modal strain response to noise takes the form of a sinusoidal variation with randomly varying amplitude. This condition is related to the constant amplitude sinusoidal siren test by Miner's Cumulative Damage Hypothesis which is known to be only approximately true. Thus even in the true uni-modal response case the extrapolation to aircraft lives cannot be exact. In cases where the panel on the aircraft will vibrate in a multi-modal form then the extrapolation from discrete to random cannot be carried out other than by empirical methods such as those put forward by Belcher. The Langley results (Figs.43(a) and (b)) illustrated this difficulty when comparison was made between random (by air jet) and discrete-frequency testing. These results on single panels cannot be used to derive general conversion factors because the proportion of energy in the higher modes than the fundamental will vary with the type of aircraft configuration used.

To overcome this difficulty random noise generators are being developed. There are three types currently in use as listed in Table VI (details have been given earlier in the text).

Both the first two devices give a broad band spectrum over the frequency range of interest (up to 1000 c.p.s.). Certain variations of spectrum can be made in the W.A.D.C. type by changing the rotor speed and the airflow but these are limited, and all have similar shapes. The Altec-Lansing air modulator is capable of producing any shaped spectrum within its operating range of frequencies but its intensity is lower than the W.A.D.C. device. The B.B.N. installation using two sirens produces a band of frequencies 50 c.p.s. wide which would certainly be wide enough to produce random response in a single mode of vibration. It would not in general, however, be capable of producing multi-modal response. The installation for the W.A.D.C. Aircraft Laboratory is to consist of twelve of these B.B.N. paired units and thus at great cost it will be able to produce broad-band high-intensity noise.

As can be seen from Table VII, the location of the panel relative to the sound wave from the siren remains a controversial question. Some feel that in practice an aircraft panel experiences primarily grazing incidence excitation from an exhaust whilst others maintain that reflections from the runway and aircraft structure result in mixed or nearly normal impingement. Additional factors generally enter into the choice of configuration because with the panel close to the horn in the normal incidence case pressure reinforcement due to reflection takes place and results in a higher-intensity loading on the panel for a given siren condition. This however has the disadvantage that in some cases during the initial frequency sweep a coupled panel/horn resonance may be picked out instead of a pure panel resonance. The grazing incidence installation on the other hand obviates this coupled resonance but loses the intensity-increase advantage. In this case larger structural specimens can generally be incorporated and it is often possible to test panel groups, not just single panels. In the grazing incidence case it is possible to excite modes having an even number of half wavelengths along the span as well as odd numbers but in the normal incidence case it is only possible to excite modes having odd numbers of half wavelengths.

The question of which is the more realistic representation really rests on the question of pressure correlation over the panel. Figure 46 shows typical narrow-band pressure correlations on a structure due to a jet efflux and also due to boundary-layer fluctuations for a central frequency of 500 c.p.s. With a normal incidence facility the siren pressure correlation is unity in both directions over the whole area of the test section. With the grazing incidence configuration pressure correlation is approximately unity in the lateral direction and takes the form of a cosine wave in the longitudinal direction. In the jet-noise case it is seen that the lateral correlation is high over 24 in. (curves are symmetrical about zero separation) and is well represented either by normal or grazing incidence layouts. In the longitudinal direction the jet-noise pressure correlation varies with position near the jet efflux and in some cases the distance to the first zero crossing point would be less than the 11 in. shown and in other cases it may be greater. With the possible exception of positions very close to the boundary it will never be less than the 6½ in. of the siren case. By orientating the specimen relative to the sound wave the 6½ in. length can be increased up to infinity (representing normal incidence) as proposed in the larger working section of the North American facility.

The boundary layer case shown is for an aircraft speed of 850 ft/sec. The spatial scale appears to increase in direct proportion to forward speed and thus will approach the siren case at supersonic speeds. In the lateral direction, however, the length over which the pressures are correlated is likely to be relatively small - of the order of a boundary-layer thickness. Thus the siren simulation, where pressures in the lateral direction are in phase over the whole of the test specimen, is likely to be unrealistic. The error involved will depend on the ratio of the lateral correlation length to the structural wavelength in this direction.

From these curves it can be seen that neither siren configurations approach closely the actual pressure correlation patterns in general although in several cases the jet-noise distribution may be reasonably well reproduced by the grazing incidence layout. The degree to which the pressure correlations must be reproduced will depend on the structural wavelength of the particular mode of vibration being investigated. If the actual pressure correlation pattern matches approximately the structural mode

pattern then it will be essential to endeavour to reproduce the pressure correlation by grazing incidence; in other cases, where the sound wavelength is long compared with the structural wavelength, both systems should give comparable results.

The problem of representing an aircraft structure in a small-scale acoustic test rig is aggravated by the fact that the form of the random vibration taking place in the actual aircraft or missile is not known. From strain gauge measurements Douglas and others have concluded that in a typical fuselage skin panels appear to vibrate in a single mode between the frames which are relatively widely spaced (about 20 in.). In other cases when the frame spacing is reduced, as in the Caravelle fuselage and control surfaces, strain gauges show a complex multi-modal response. In the multi-modal case it is clearly insufficient to consider only a single panel vibrating in its fundamental mode, and therefore much more extensive spatial groupings of panels and support structure must be used. As a result of this the tendency in most of the siren-test facilities visited was to move on from the single-panel testing, of which a lot was done in the early days, to tests on panel groups and sections of flaps, etc. This trend reaches its limit in the W.A.D.C. siren facility, now being constructed, where whole wings or large sections of fuselage could be placed near the sirens. Over large areas of structure like this, however, the siren may not reproduce the actual pressure correlations and, indeed, unrealistic forms of vibration may be set up. As the form of vibration taking place appears to be relatively local rather than overall it is difficult to see what additional significant information can be gained from this type of test as opposed to a programme using single sirens and smaller structural sections.

In small-scale tests the type of panel mounting has a considerable effect on fatigue life. In the case of single panels most companies have used fully fixed-edge conditions, partly to get repeatable test results and also because this was considered to be more representative of the aircraft structure. Figure 47 shows a comparison of the effect of three different types of edge condition. The simple bolted edge gave least scatter but the lowest life, as failure started at the bolt hole. In the bonded cases the stress concentrations are reduced and the mean life increased but scatter is also increased. Thus the simple bolted configuration is used in many cases. Where multi-panel groups or flap sections are being tested the edge conditions are not quite so critical. In these cases the structure can be mounted from the ribs or frames, care being taken to attach the edges to prevent flapping.

3.3.7 Relative Merits of Structural Configurations

It is now clear that the life of a particular piece of an aircraft structure which is being subjected to acoustic loads depends on a number of parameters, many of which are only qualitatively understood. In these circumstances it is not possible to formulate rigid design rules which will ensure the integrity of any particular type of structure. However, as a result of considerable experience, it is possible to point out one or two improvements. Loose edges such as might occur on a control surface nose rib should be avoided and plain rivet holes are much better than dimpled or cut countersunk. Attempts to produce design charts for skins and rib pitches have been made by Douglas and de Havillands but these are based on particular aircraft and have been found to be inapplicable to the structure on other aircraft.

For basic skin subjected to high-intensity noise pressures tests have shown and service experience now confirmed that honeycomb sandwich structures are able to withstand higher loads than an equivalent single metal skin. The noise level at which it becomes necessary to use honeycomb instead of sheet skin is not clear but it is likely to be in the region of 160 db. The majority of failures which have troubled the aircraft manufacturers have been in detail attachments. The experience of several companies now confirms that in order to avoid failures in rib flanges it is necessary to use a symmetrical form of rib-skin attachment. De Havillands achieved a test life increase of over 1000 times by using back-to-back doublers at the rib/skin joint. Boeing quote a service life of under 100 hours with a normal pressed over rib flange and over 2000 hours with a symmetrical joint. Douglas have successfully developed an X cross-section extrusion for use in attachment of stringers to frames or ribs as shown in Figure 48.

3.4 CONCLUSIONS

The main conclusion to be drawn from this section is that, in spite of considerable effort, no completely satisfactory method of allowing for acoustic fatigue is available. It is apparent, however, that by processes of trial and error the majority of companies have developed structure which is satisfactory for acoustic loading. In so doing they have gained a wide experience of the types of failure which are likely to occur and have developed semi-empirical methods of design.

The comprehensive theoretical approaches of the normal mode type at first sight appear to involve a prohibitive amount of computation. Experimental investigation to determine the types of mode likely to be excited by jet noise should lead to a reduction of the number of modes which need be considered. The amount of labour involved in the wave-theory approach is unlikely to be significantly less, but it would seem to be worthwhile to pursue this method further to ascertain its possibilities.

It is clear that lack of modal information is also holding up advances in other aspects of the problem. The study of damping is rapidly reaching the stage where both rivet and acoustic damping could be calculated if the mode of vibration were known. Similarly, the effectiveness of the representation of the structure in a test rig depends on an adequate representation of boundary conditions. More extensive modal information of the type obtained on the Caravelle rear fuselage will enable more realistic sizes of structural test pieces to be determined. For example, in the Caravelle case the mode information suggests that an adequate test specimen would consist of a long strip of skin and stringers supported between two frames. In order to get the frame conditions right three frame bays might be used.

The simplified theories of response are not adequate to deal with practical structures in terms of relating life to noise level. They do provide, however, the basis of reasonable engineering methods of relating siren test life to aircraft service life. More development of the Douglas method on an experimental basis would seem to be well justified. The effect of multi-modal response should be investigated on simple specimens.

From strain response data it is clear that a resonant response is taking place and that anything from one to six or more major resonances may be excited by noise. The frequency range is 100 to 1000 c.p.s. with conventional structure. The only general

conclusion is that a skin panel on relatively widely spaced frames (20 in.) will usually have a uni-modal response. In most other cases a multi-modal type of response occurs.

A few isolated measurements have shown that strain attenuation does occur when noise suppressors are fitted to the jets. Also at take-off the strain level falls rapidly once the aircraft has started to move along the runway. The rate at which strain falls off with forward velocity will depend on the position of the structure relative to the jet. Only along the jet boundary, and close to the nozzle (within about 3 diameters), is there likely to be little fall-off in strain with forward velocity.

The most difficult aspect on which to draw any general conclusions is the question of test facilities. There appear to be almost as many test philosophies as there are aircraft companies. It is clear, however, that any claims to be able to design structure accurately on siren tests alone cannot be substantiated. It is equally true that more information can be obtained than just which of a group of panels is the 'strongest'. The Douglas approach taking into account non-linearities and multi-modal response with a third factor added for mounting conditions offers the most realistic way of using siren data. Even in this case tests on an engine are the final step.

SECTION IV

AREAS FOR FURTHER RESEARCH

Many of the topics requiring further investigation are included in the research programmes of the various establishments; nevertheless, it is clear that several aspects need additional work. A list of suggested projects is given below.

Noise Levels and Spectra

(1) Theoretical study of possible extrapolation procedures for near-field noise of jets and rockets.

(2) Measurements of the effect of structural reflections, suppressors and thrust reversers and comparison with free-field noise.

(3) Measurement of noise levels and spectra forward of efflux and adjacent to intakes.

(4) Measurement of levels inside nose ribs of flaps and elevators.

(5) Full-scale correlation measurements on a jet.

Boundary Layer

(1) Tunnel wall levels and spectra in Mach number range 1.5 - 5.

(2) Filtered space correlations on the wall at speeds from 500 ft/sec upwards.

(3) Further flight measurements to cover a wider range of Mach numbers and boundary-layer thicknesses.

Structural Response

(1) Theoretical development of wave method.

(2) Theoretical development of energy method.

(3) Normal modes of stiffened shell structures.

(4) Construction of a stiffened cylinder (with uniform stiffener spacing) and later a honeycomb cylinder; resonance tests followed by excitation by jet noise to evaluate response.

(5) More extensive investigation of types of modes being excited on actual structures.

(6) Application of damping materials to practical configurations.

(7) Effect of multi-modal response on life of simple specimens.

(8) Further development of the siren test procedure.

(9) Strain amplitude distribution - further check on occurrence of isolated peaks by electronic counter.

Isolation of Guidance and Control Equipment

A detailed review of present knowledge is required.

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TABLE I

Errors in Prediction of Near-Field Acoustic Loads (Ref. 31)

Method of prediction	*Error* (db)	*Error* (%)
Extrapolation of free-field data from 10,000 lb thrust engine to structural loads for 20,000 lb engine	±7	+124% - 55%
Extrapolation of free-field data from 10,000 lb thrust engine to structural loads for 15,000 lb engine	±5	+ 78% - 44%
Extrapolation of measured acoustical loads for thrust variations in the order of 2000 lb	±3	+ 41% - 29%
Interpolation of measured acoustical loads between thrust levels of 10,000 lb and 12,000 lb	±2	+ 26% - 21%
Measurement at a point on the aircraft structure	±1	+ 12% - 11%

TABLE II

Comparison between Measured and Theoretical Results (see Sec. 3.2.16)

Mode type	*Motion of neighbouring panels*	*Frequency c.p.s.*	
		computed	*observed*
(a) Symmetrical about centre of span	Out of phase	104.5	102
(b) Symmetrical about centre of span	In phase	414.0	420
(c) Antisymmetrical about centre of span	Out of phase	232.3	237
(d) Antisymmetrical about centre of span	In phase	496.6	490

TABLE III

Comparison of Theoretical Computations for Interframe Mode Frequencies

Structure	*Experimental*	*Lin*		*Energy (Rayleigh-Ritz)*	
		without stringers	*complete*	*without stringers*	*complete*
Caravelle					
Panel 2(a)	590	586			724
5(a)	720	626			727
5(b)	?	1067			1067
Boeing 707					
Mode (a)	102	104.5	119.5	106.8	121.4
(b)	420	414.0	415.1	414.3	414.8
(c)	237	232.3	229.2		
(d)	490	496.6	497.5		

TABLE IV

Overall Damping Present in Untreated Structures

Structure	*Frequency* (c.p.s.)	*Damping ratio*
Caravelle rear fuselage skin panel	600 - 700	0.016 - 0.018
Comet tailplane skin panel	400	0.014
Douglas test panel 24 in. × 24 in.	120	0.022
Douglas test panel 24 in. × 24 in.	320	0.019
NACA Langley test panels 11 in. × 13 in., thicknesses 0.04 - 0.081 in.		
(a) without sound absorbent backing	100 - 1000	0.012 - 0.02
(b) with sound absorbent backing	100 - 1000	0.032 - 0.05

TABLE V

Major Frequencies of Strain Spectra

Aircraft structure	*Frame or rib pitch* (in.)	*Stringer pitch* (in.)	*Major frequencies* (c.p.s.)
Douglas RB-66 Control surface skin Fuselage stringer Rib flange	24 4	4 - 6	230 400, 470, 540, 600 200, 230, 270 360, 470, 540
Caravelle rear fuselage skin	8	4	700 (850, 940)
Comet tailplane, skin stringer crown	12	6	380 280, 340, 380, 440
Boeing fuselage skin	20	8	100, 400, 470

TABLE VI

Summary of Siren Facilities

Facility	*Type*	*Spectrum*	*Level db*
WADC (Aeromedical Lab).	Multi-rotor	Fixed 200-10,000 c.p.s.	165
Boeing, Seattle Southampton Univ. (Altec-Lansing)	Electro-magnetic air modulator	Variable 100-1000 c.p.s.	150
B.B.N.	Two-component narrow band modulated	50 cps band width centred in range 50 - 10,000 c.p.s.	170

TABLE VII

Details of some Siren Test Facilities

Facility	*Sound level (db)*	*Frequency range (d.p.s.)*	*Plenum chamber pressure (lb/in.²)*	*Air flow (ft³/min)*	*Test section dimensions*	*Angle of incidence*
Douglas, Santa Monica	163	50 - 1000	5	4080	Up to 4ft × 5ft	Grazing
Convair, San Diego	170	50 - 1000	2	1800	22in. × 44in.	Grazing
Convair, Fort Worth	170	150 - 1000	10	2000	"	Normal
* Convair, Fort Worth	170 - 180	<100 - 2500	10	~6000	"	Normal
Boeing, Wichita	160 - 170	- 1000	5 - 40	1200	"	Normal
Boeing, Seattle	170	100 - 650	50	1400	"	Normal
" "	170	100 - 950	50	1400	"	Normal
" "	150	Random variable spectrum 100 - 800	25	150	6ft × 6ft	Reverberant/ anechoic
* North American, L.A.	170	50 - 10,000			3ft × 5ft	Grazing
* W.A.D.C. Aircraft Lab.	160	Random or discrete 50 - 10,000			70ft × 50ft × 42ft 50 sq. ft. floor area	Reverberant grazing or normal
" "	174	Random 50 - 10,000				
W.A.D.C. (Aeromedical)	165	Random fixed spectrum 200 - 10,000				
Southampton University	150	Random variable spectrum 130 - 800	40	200	2ft × 3ft	Grazing
* Vickers Supermarine	165	100 - 2000	80		3ft × 3ft	Normal

* Under Construction

TABLE VIII

List of Establishments, Investigators, Facilities and Programmes

Establishment	*Investigators*	*Facilities*	*Details of work*
EUROPE			
1. R.A.E.	M.O.W. Wolfe W.T. Kirkby D.R. Webb C. Skingle	Open-air test rigs Ghost & Avon RA.26 with re-heat Rocket engine (scale models)	Near-field levels and spectra, some correlations. (Main programme of tests completed). Structural response: sections of structure and panels close to jets. Noise levels in launching tubes. Boundary-layer pressures (and panel response) on Fairey Delta.
2. Rolls-Royce	H. Spooner G. Coles D. Middleton	Open-air test bed Palas 600 scale Engine Model hot air jet (high pressure ratio) I.B.M.	Near-field noise measurements on full-scale jet engines. Some near-field work on 8 corrugated nozzles. Computer programme for pressures on structure.
3. De Havilland	N.A. Townsend	Tail plane test sections (with R.A.E.) Fatigue tests on joints	Stress levels & fatigue life behind engine at R.A.E. S-N curves for low stress, long life end of curve. Design of peak counter for fatigue life estimation.
4. Vickers	D.C. Thomas	Discrete-frequency, normal incidence siren (under construction)	Fatigue tests of panel groups T/P proof tests at R.A.E.
5. Imperial College	C.G. Parfitt	Vibration test equipment	Damping of constrained layers.

TABLE VIII (continued)

Establishment	*Investigators*	*Facilities*	*Details of work*
6. Southampton University	E.J. Richards D.J. Mead B.L. Clarkson M.K. Bull I. Torbe T.R.G. Williams	Model air jets 1in. to 4in. dia. Correlator (DC-20,000 c.p.s.)(time delay 2μs, 140 ms). Loudspeaker unit Grazing incidence, random siren. High-intensity resonance tube, Vibration Lab. Random fatigue testing rig	Theoretical & experimental study of noise production by jet. Noise pressure correlation. Strain correlation. Resonance test of large sections of structure. Assessment of siren test techniques, crack propagation. Fatigue life of honeycomb panels. Damping due to (i) structure rivets (ii) accustic rad., (iii) Added materials. Efficiency of damping.
7. College of Aeronautics	G.M. Lilley T.H. Hodgson	Wall jet	Theory of jet noise. Theoretical and experimental study of boundary-layer pressure.
8. O.N.E.R.A. (Paris)	E.A. Lienard M. Kobrynski C. Contour	Mabore open-air test bed Atar 101F	Test rig for panels; strain distrib: damping, fatigue. Near-field levels and spectra.
9. Sud Aviation (Toulouse)	J. Wagner Chaussonnet	Rear fuselage and tail unit of Caravelle + 1 engine	Strains, fatigue life. Noise pressures on surface of structure (Kobrynski, O.N.E.R.A.)
10. D.V.L. (Mulheim)	W. Thielemann P. Waldow	Random fatigue rig	Effect of waveform on fatigue. Stability of thin shell structures.
11. Gottingen	F.A. Müller	Turbulent wind tunnel	Scattering of sound by turbulence.
12. University of Rome	L. Broglio Santini		Dimensional analysis of jets. Random excitation of structures. } To be started

TABLE VIII (continued)

Establishment	*Investigators*	*Facilities*	*Details of work*
13. Polytechnic of Turin	C. Ferrari	Mach 2.5 wind tunnel	Boundary-layer turbulence.
14. Armstrong Whitworth	Melrose	Vibration test equipment	Noise measurements on missile.
15. Cementation (Muffelite)	J.W. Gearing		Tests on vibration isolators.
<u>U.S.A. AND CANADA</u> 1. N.A.S.A., Cleveland	J.C. Fakan H.R. Mull J. Seraphini	Model and full scale jets Subsonic/supersonic wind	Noise measurements. Boundary-layer pressures; levels, spectra and correlation
2. N.A.S.A., (Langley)	A.A. Regier G.W. Brooks G. Kantarges H. Hubbard W.H. Mayes H.F. Hardrath	Model jet facility (hot) 12in. diam. high turbulence jet	Measured spectra on Atlas; Model tests Response of panel to boundary-layer pressures Panel response and fatigue life.
3. W.A.D.C.	H.E. von Gierke K. Eldred W.T. Trapp D.M. Furney Magrath O.M. Rogers	Reverberation room, Anechoic room. Random siren. Very large siren facility being designed	Noise suppressors Boundary-layer pressures measurements on F.100 and F.102 aircraft Fatigue of large items of structure and equipment
4. B.B.N.	I. Dyer P.A. Franken E.E. Ungar E.M. Kerwin	Modulated siren	Theoretical work on jet and rocket noise, and boundary pressures. Design of siren facility for W.A.D.C. Damping of constrained layers and fibreglass blankets

TABLE VIII (continued)

Establishment	*Investigators*	*Facilities*	*Details of work*
5. Minnesota University	B.J. Lazan T.J. Mentel R.F. Lambert	Damping measurements Space-time correlator Boundary-layer tunnel	Damping Fatigue Hot-wire correlation measurements
6. U.C.L.A.	Leonard Rudnick E.R. Shanley A. Powell R.C. Chanaud P.H. White H.H. Unfired	Anechoic chamber 4in. diam. air jet, flow instability apparatus	Design consultant on sirens Analysis of fatigue test data. Theoretical work on noise production and response
7. Avro, Canada	H.H. Shiji	Programme stopped	Fatigue tests using siren. Effect of damping
8. Boeing	R.H. Gibbs V.E. Callaway J. Fuller Y.K. Lin L.C. Sutherland Large	1/8th scale model jet Siren (discrete f.normal) Random siren (Altec-Lansing) Reverb/ Anechoic. Engine test cells	Evaluation of noise suppressors. Panel fatigue tests. Response of large structural specimens, sound transmission. Test cell, engines on pylon, large section of wing, fatigue tests. Theoretical work on normal modes
9. Convair	G. Getline	Siren (discrete, grazing)	Fatigue tests of panel groups and sandwich construction. Structural design procedure
10. Douglas	M.M. Miller T.J. Shultz* F.S. Meyer J.D. van Dyke A.L. Eshleman J.C. McClymonds A.H. Marsh	Hot 1/5 scale mode jet Siren (discrete, grazing) Jet engine cells Reverberation room, Anechoic room, open air stand with jet engine, rocket test stands	Evaluation of noise suppressors. Panel group tests and development of structural design procedure. Acoustic proof testing, sound transmission loss, static firings of rocket motors

* (Now with B.B.N. - 1961)

TABLE VIII (continued)

Establishment	*Investigators*	*Facilities*	*Details of work*
11. Lockheed	J. Rebman J.B. Vignot	Open air test stand with jet engine	Near-field levels, spectra and correlations Panel response
12. M.M.M.	C. Dalquist	Machine to measure basic properties of damping materials	Development of damping materials
13. North American (L.A.)	P.M. Belcher W.B. Greenwood	Siren (discrete, grazing) under construction	Model tests 1/10 scale hot jet, rigid structure to give pressure dist. on structure due to multi-jet layout. Panel and equipment tests in siren (planned)
14. U.T.I.A.	H.S. Ribner	Low speed wind tunnel	Noise due to structural vibration excited by boundary layer
15. Columbia University	A.M. Freudenthal R. Hel W.L. Nelson	Random fatigue machines (bending) Rotating cylinder	Fatigue under random loading. Effect of distribution r.m.s. pressure, frequency spectra
16. David Taylor Model Basin	M. Harrison G.J. Franz	Low-speed wind tunnel	R.m.s. pressure. Frequency spectra. Longitudinal space-time correlations. Convection speeds. Transverse space correlations R.m.s. pressure. Frequency spectra
17. Ordnance Research Lab., Pennsylvania State University	E. Skudrzyk G. Haddle	Rotating cylinder. Garfield Thomas water tunnel. Buoyant body	R.m.s. pressure. Frequency spectra Roughness effects
18. California Institute of Technology	W.W. Willmarth	Subsonic wind tunnel (circular pipe)	R.m.s. pressure. Frequency spectra. Longitudinal space-time correlations. Convection speeds

TABLE VIII (continued)

Establishment	*Investigators*	*Facilities*	*Details of work*
19. Jet Propulsion Laboratory	J. Laufer A. Kistler	Supersonic wind tunnel	R.m.s. pressure. Frequency spectra. Longitudinal space-time correlations. Convection speeds
20. N.A.C.A., Edwards Air Force Base, California	N.J. McLeod G.H. Jordan	B-47A aircraft D-558-II aircraft	R.m.s. pressures. Frequency spectra Internal sound levels Internal sound levels

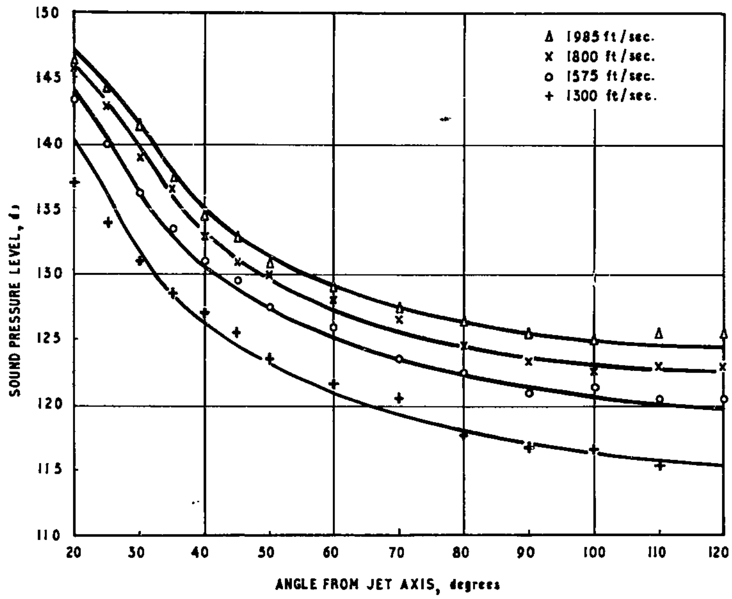


Fig.1(a) Near-field noise contours, 10 ft radius, 150 - 600 c.p.s. band (Ref.5)

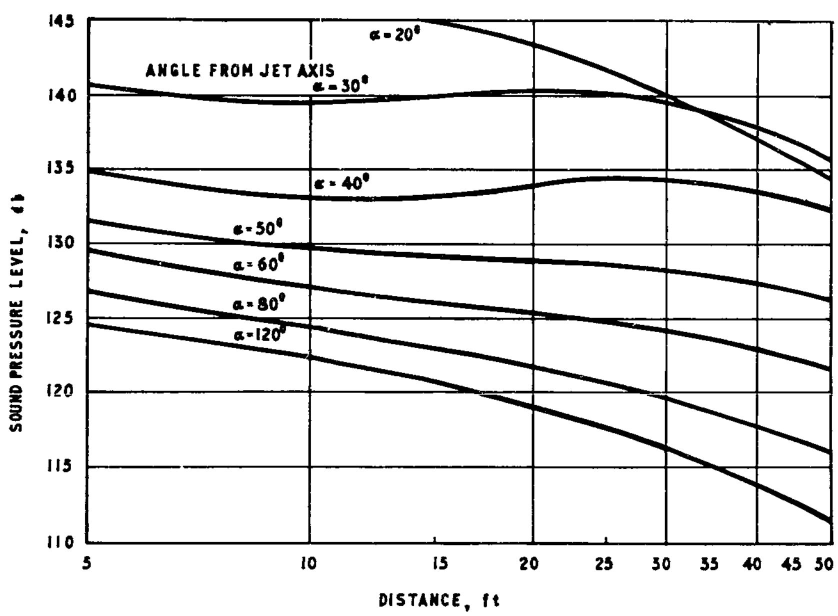


Fig.1(b) Near-field noise contours, 1800 ft/sec 150 - 600 c.p.s. band (Ref.5)

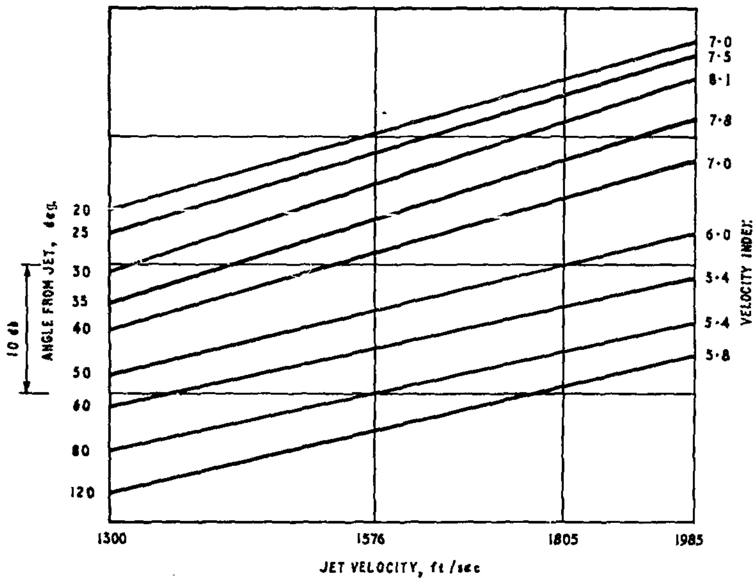


Fig.1(c) Velocity indices, 20 ft radius, 150 - 600 c.p.s. band (Ref.5)

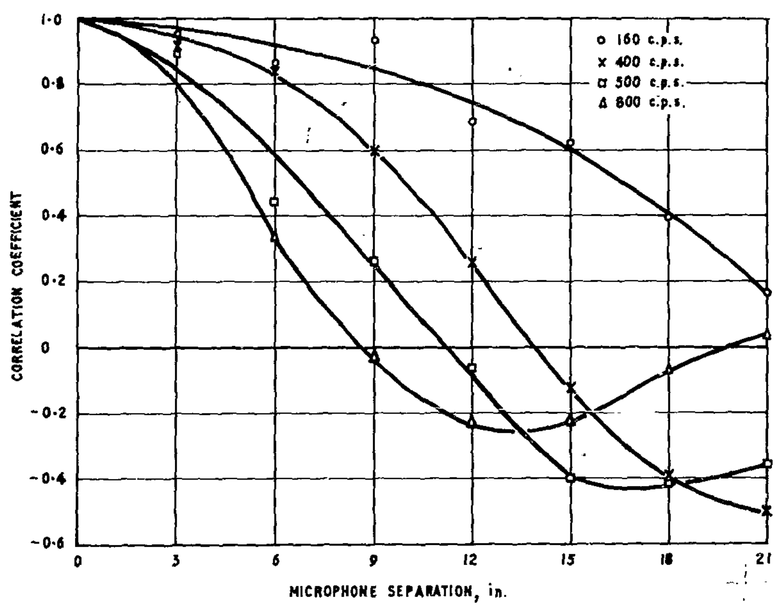


Fig.2(a) Filtered space correlograms of noise pressure on a tailplane surface (longitudinal) (Ref.64)

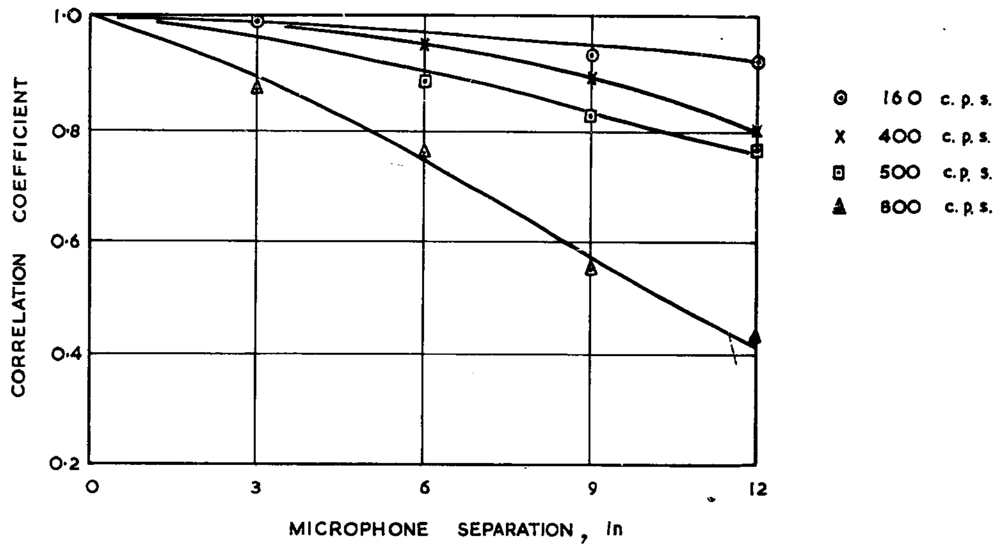


Fig.2(b) Filtered space correlograms of noise pressure on a tailplane surface (lateral) (Ref.64)

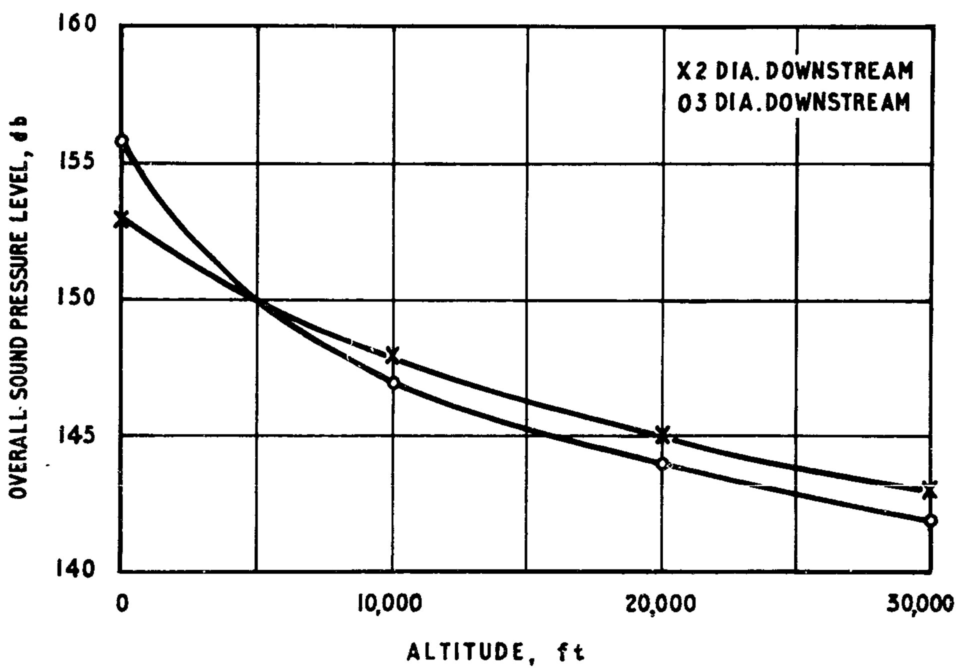
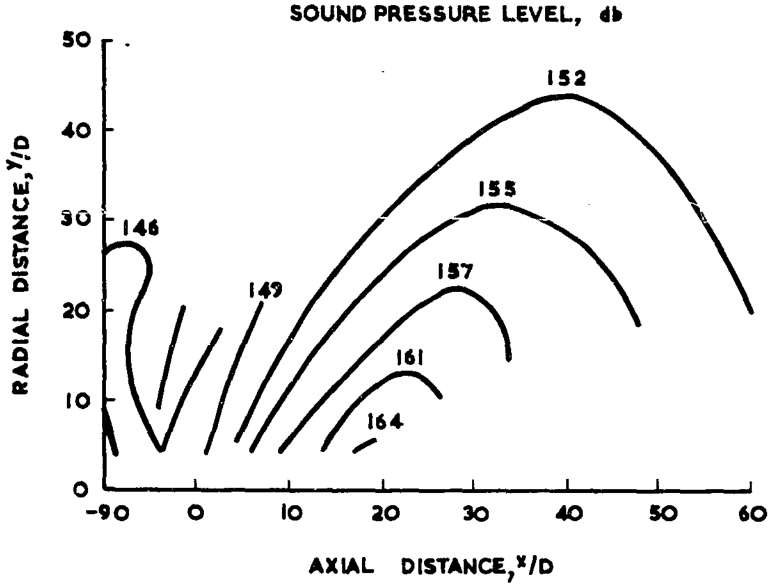


Fig.3 Variation of sound pressure level at jet boundary with altitude (Ref.15)

(a)



(b)

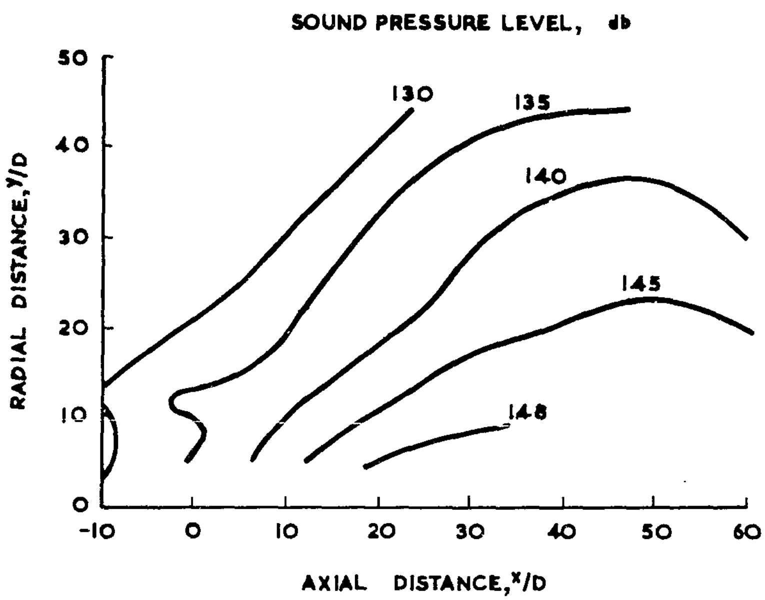


Fig.4 Contours of near-field sound pressure levels for rocket engine
(a) frequency band 5 to 2500 c.p.s., (b) frequency band 448 to 566 c.p.s.
(Ref.16)

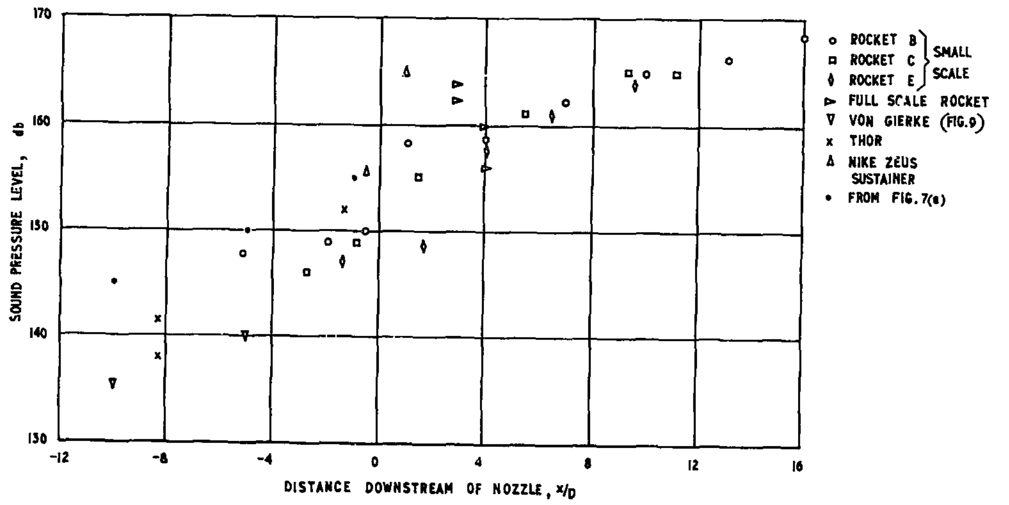


Fig.5 Variation of sound pressure level with axial distance downstream of rocket nozzle

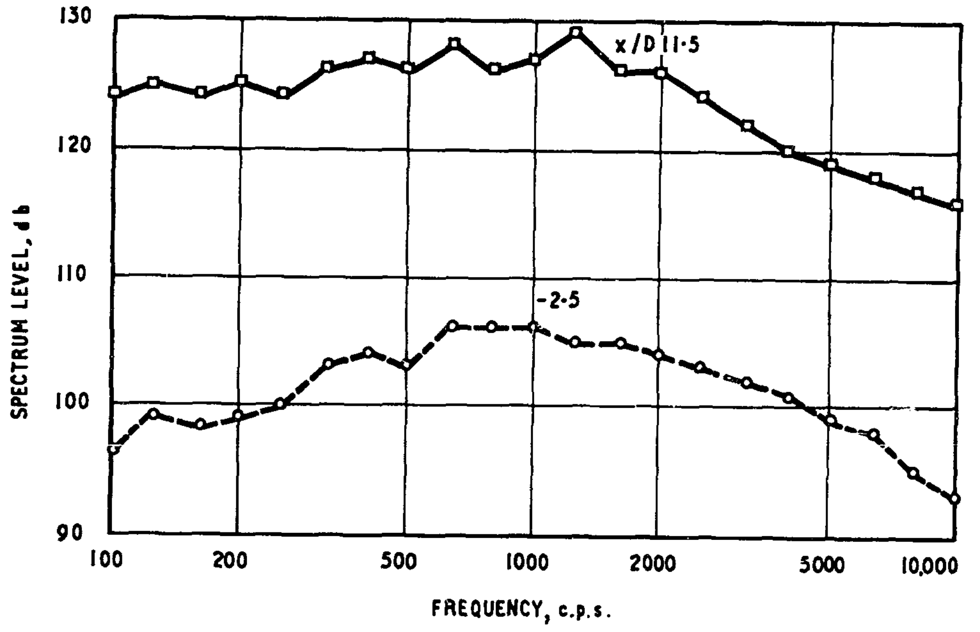


Fig.6 Near-field rocket spectra upstream and downstream of nozzle (Ref.16)

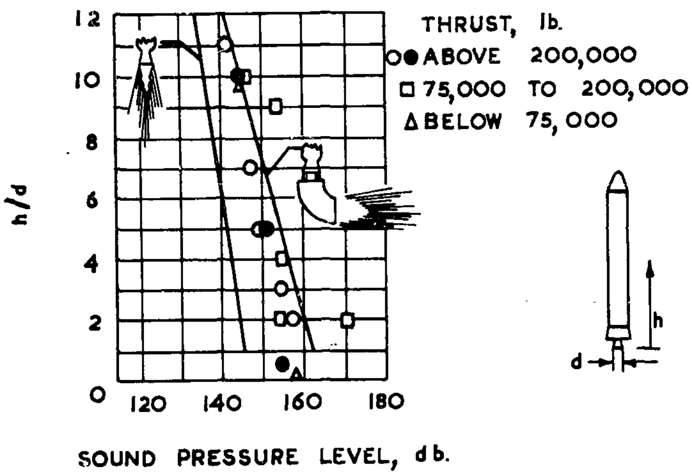


Fig.7(a) Sound pressure levels on missile surface at lift-off (Ref.17)

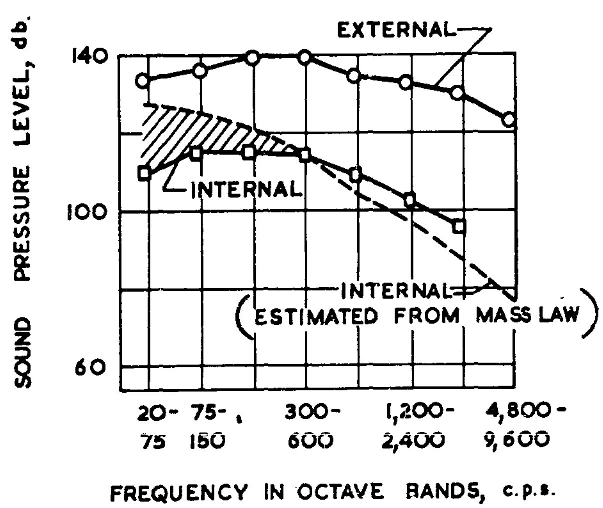


Fig.7(b) Noise spectra at Mercury capsule due to rocket (Ref.17)

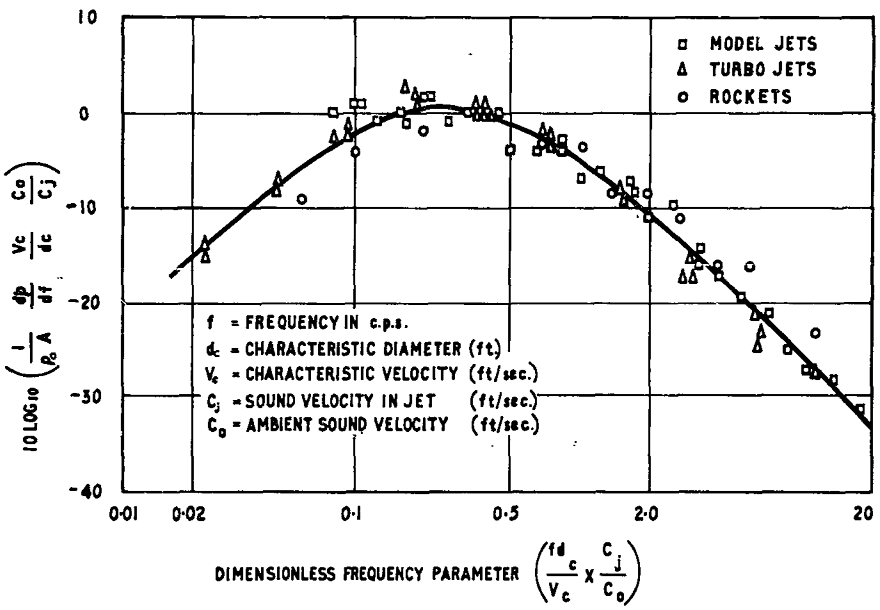


Fig.8(a) Power spectra of rockets, turbo-jets and model air jets (Ref.19)

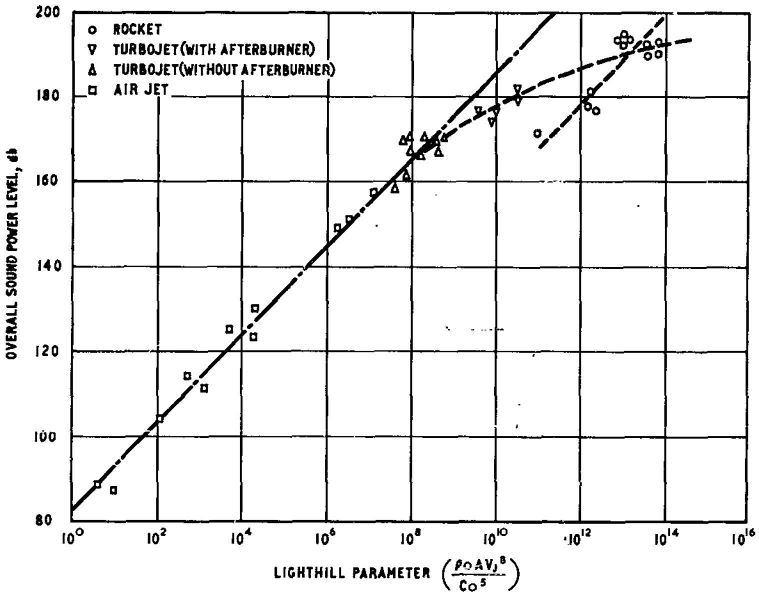


Fig.8(b) Overall sound power of rockets, turbo-jets and model air jets (Ref.19)

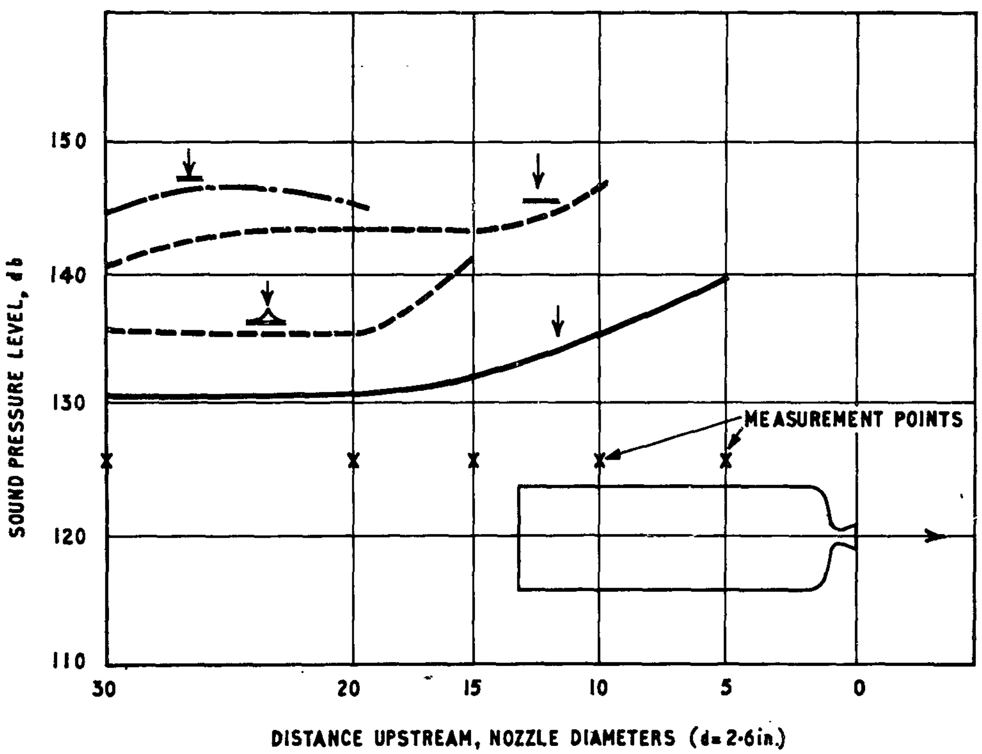


Fig.9 Near-field overall sound pressure levels for radially symmetric deflectors (Ref.19)

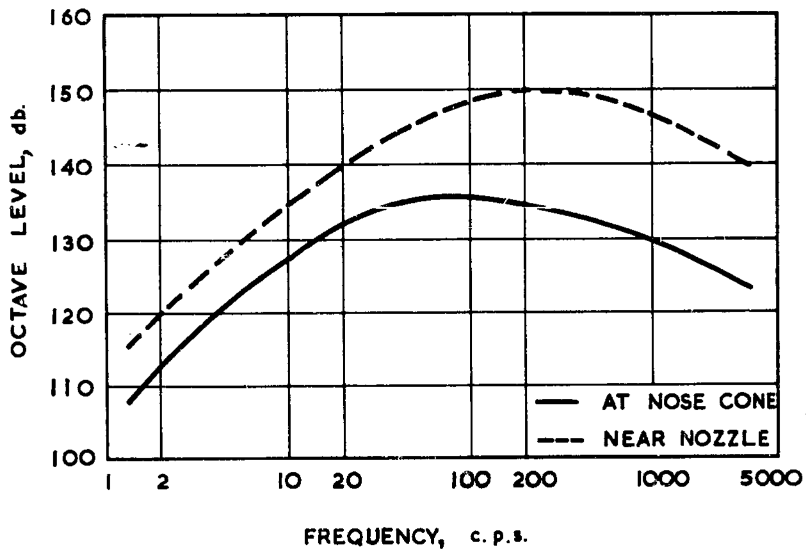


Fig.10 Sound pressure spectra on a missile surface (Ref.22)

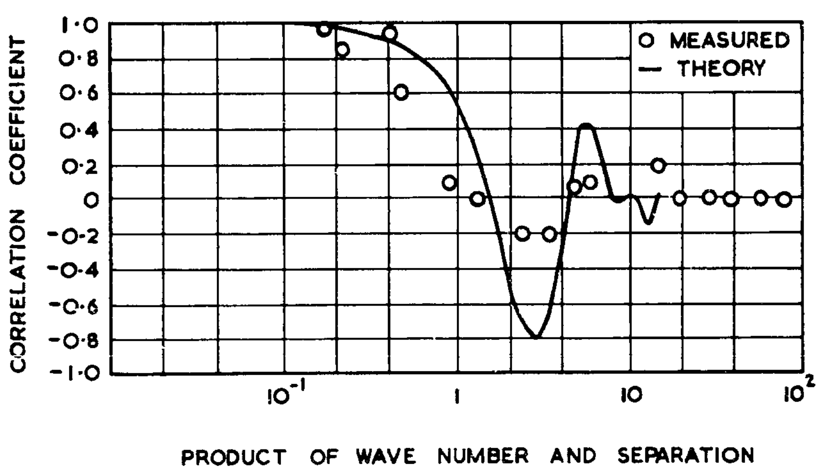


Fig.11 Octave band space correlation of the pressure on a missile surface (Ref.22)

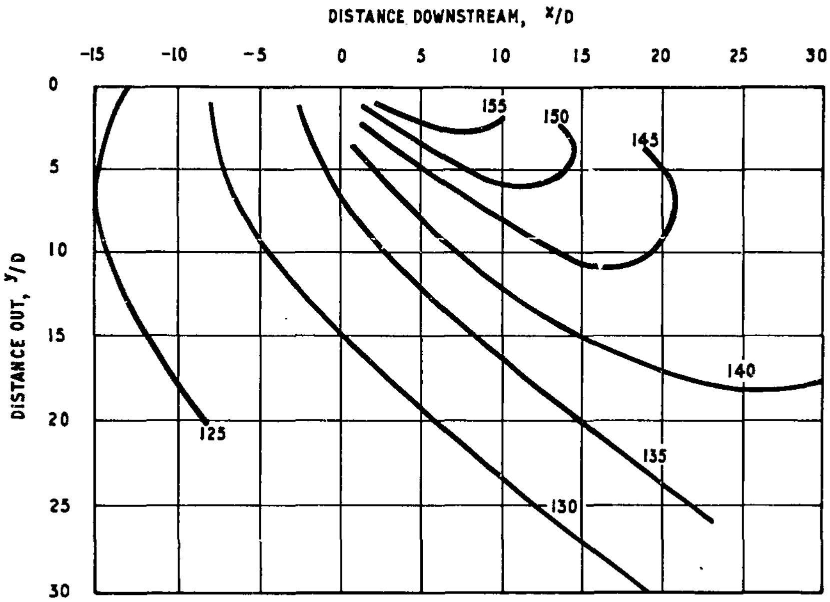


Fig. 12(a) Reference sound pressure level contours, 1850 ft/sec (Ref. 23)

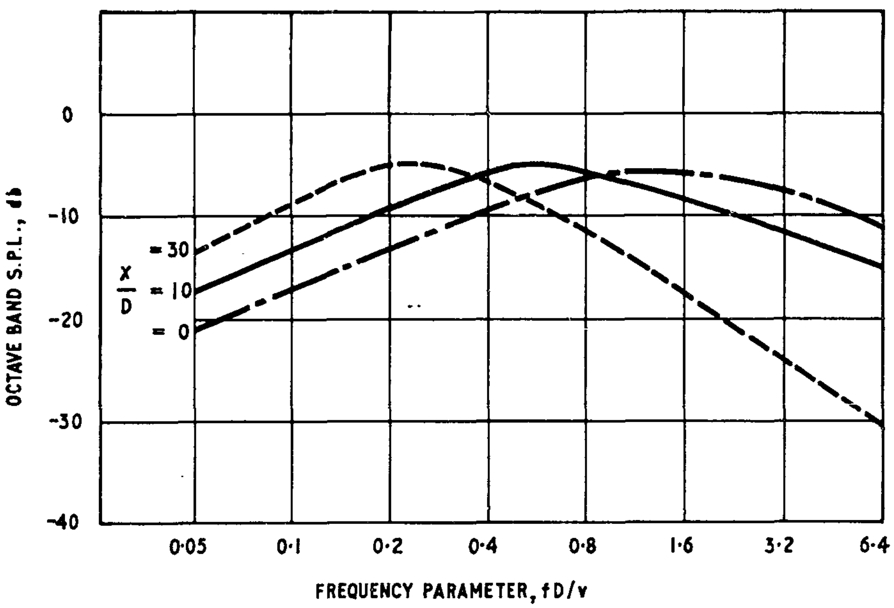


Fig. 12(b) Reference spectra (Ref. 23)

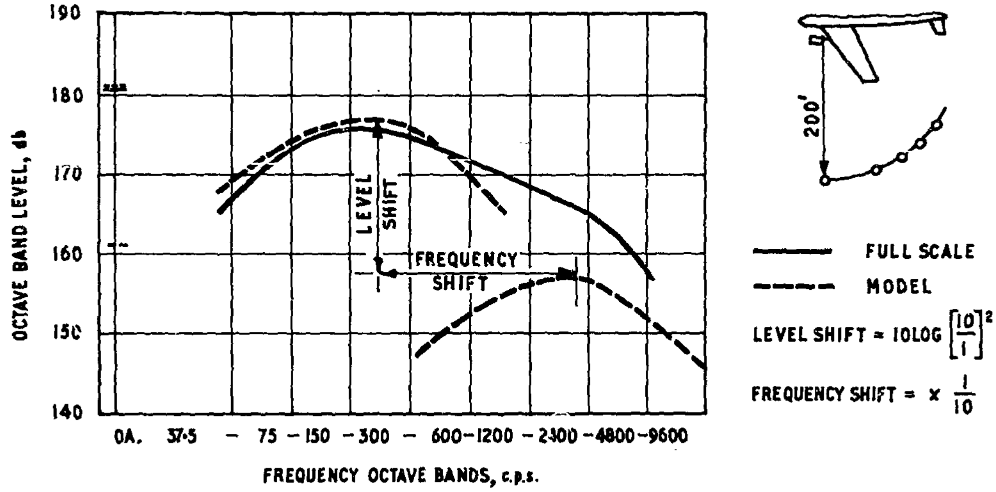


Fig.13(a) Comparison of model and full-scale spectra and levels (Ref.24)

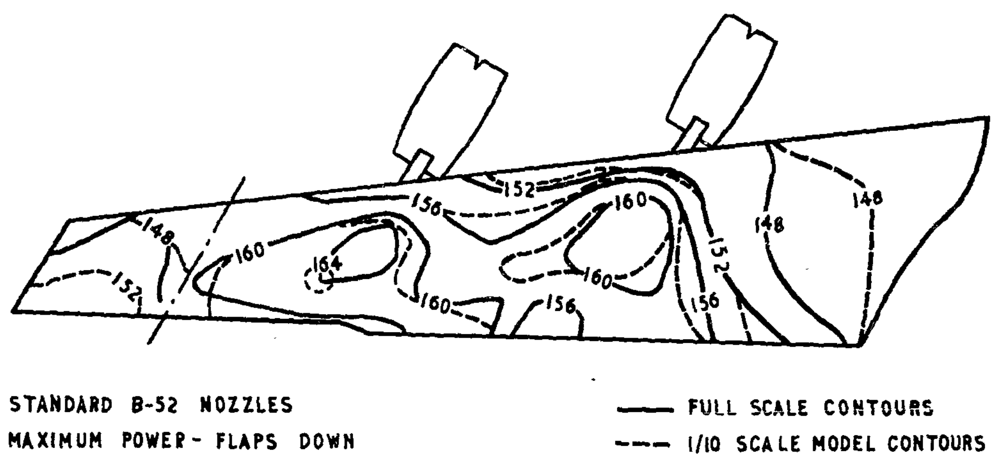


Fig.13(b) Comparison of model and full-scale B-52 sound levels (Ref.24)

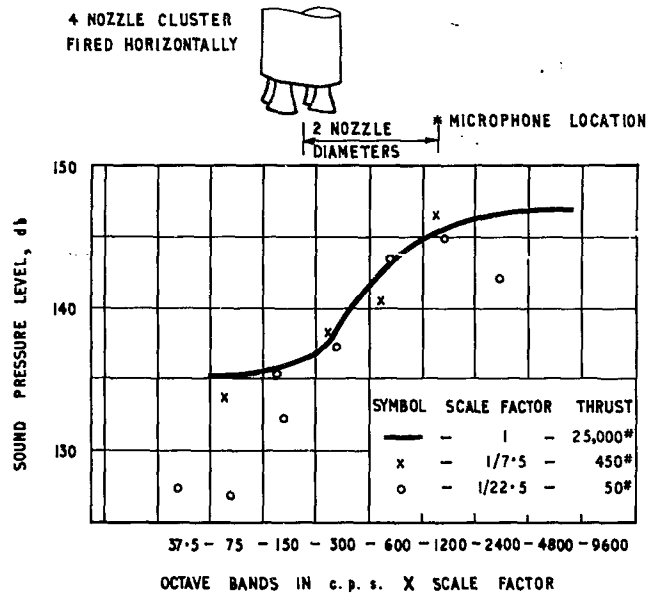


Fig.14 Comparison of model and full-scale rockets (Ref.24)

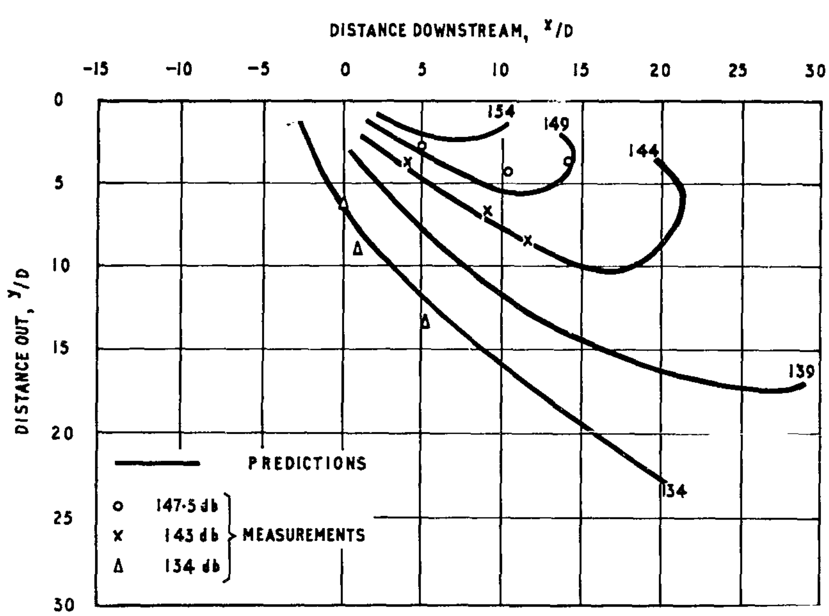


Fig.15(a) Overall level contours, 1800 ft/sec

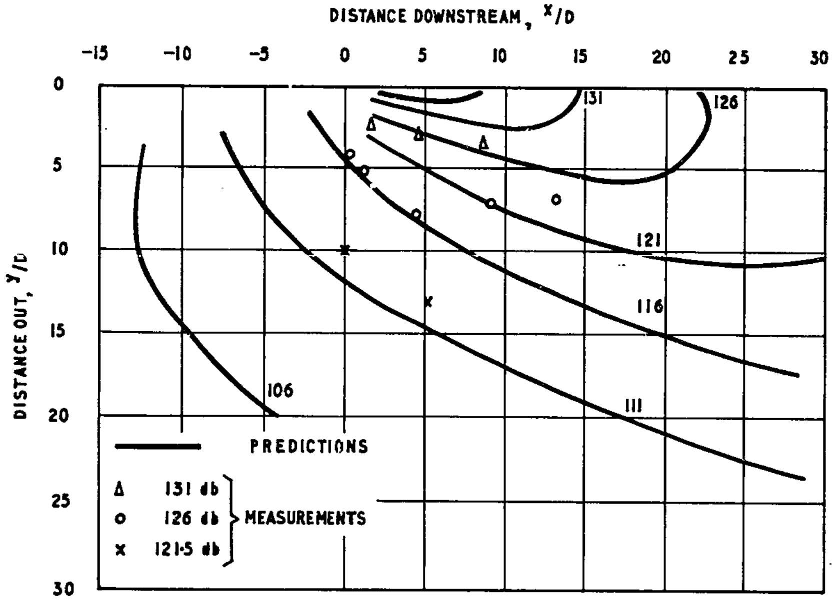


Fig. 15(b) Overall level contours, 1080 ft/sec

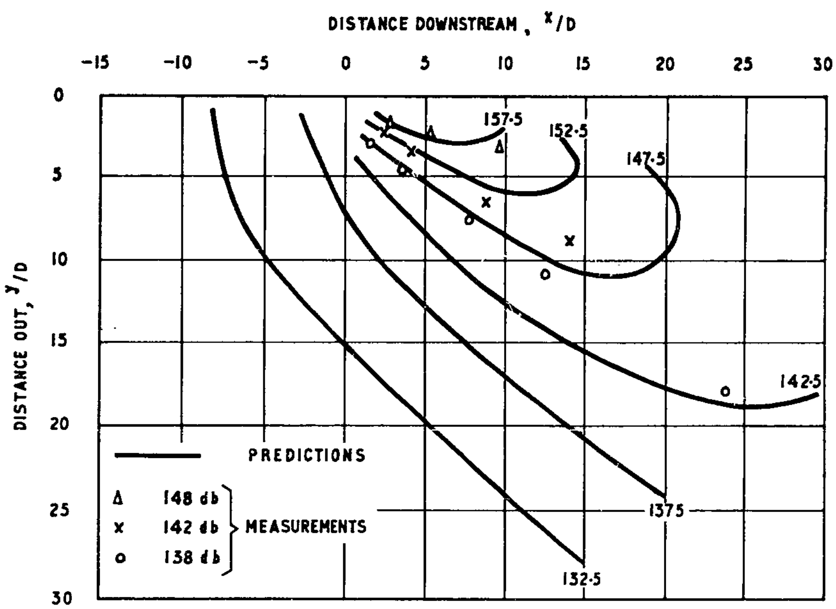


Fig. 15(c) Overall level contours, 1985 ft/sec

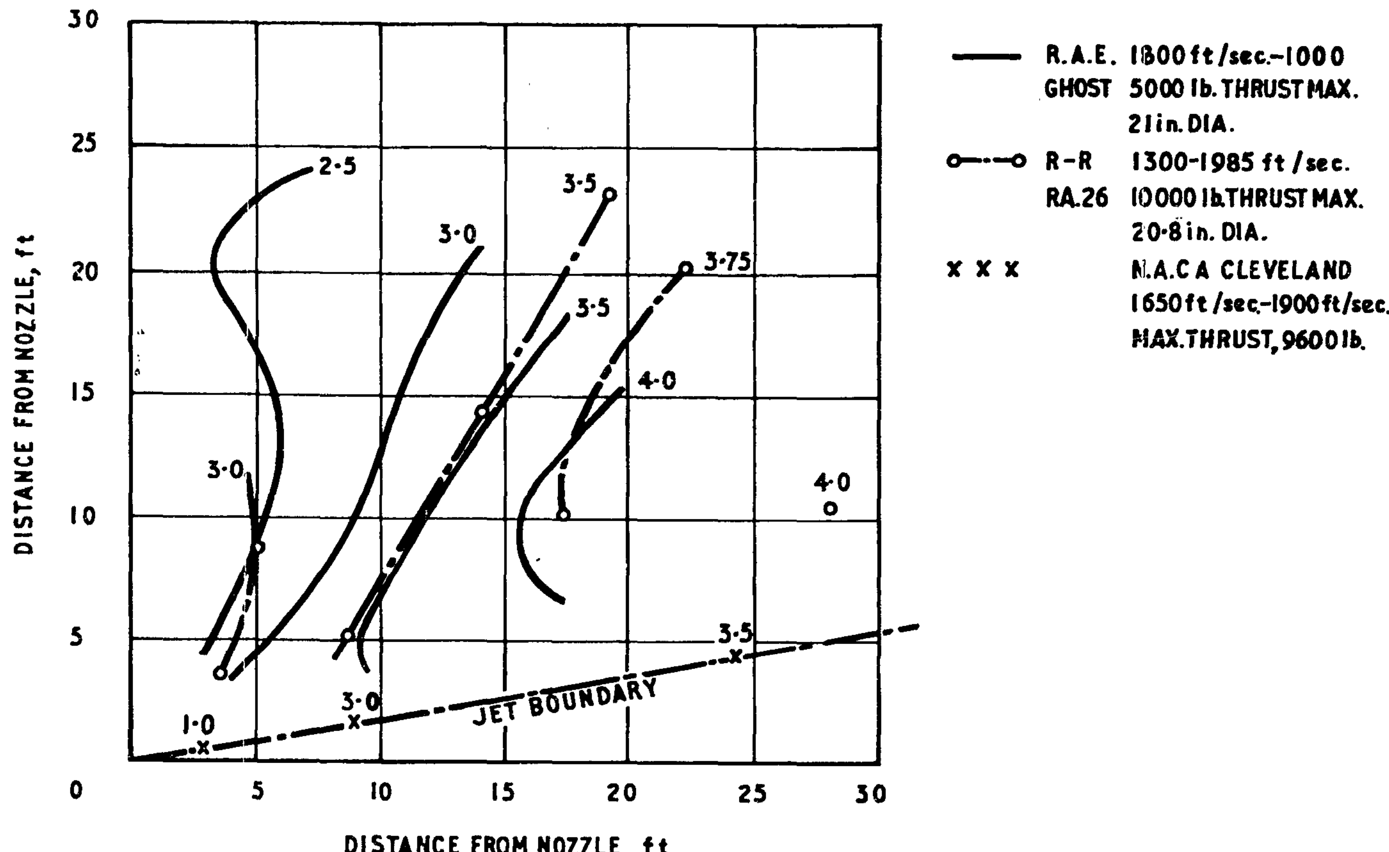


Fig.16 Comparison of near-field velocity indices

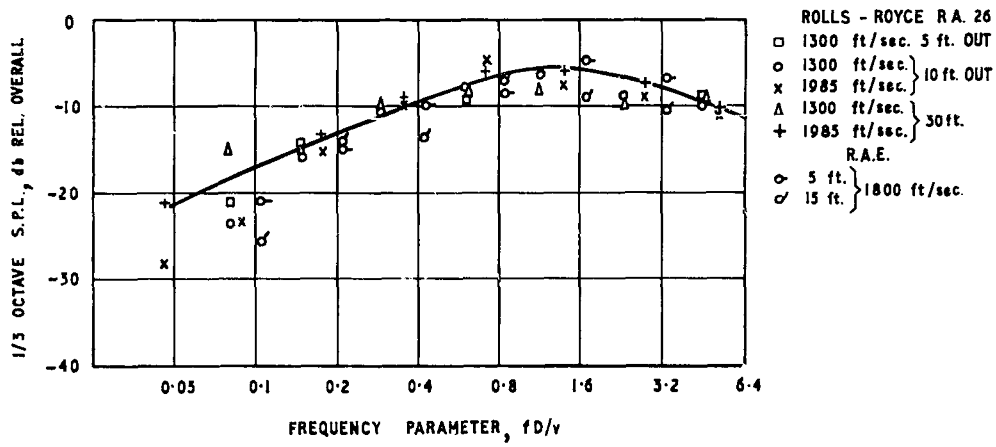


Fig.17(a) Comparison of spectra in plane of jet nozzle

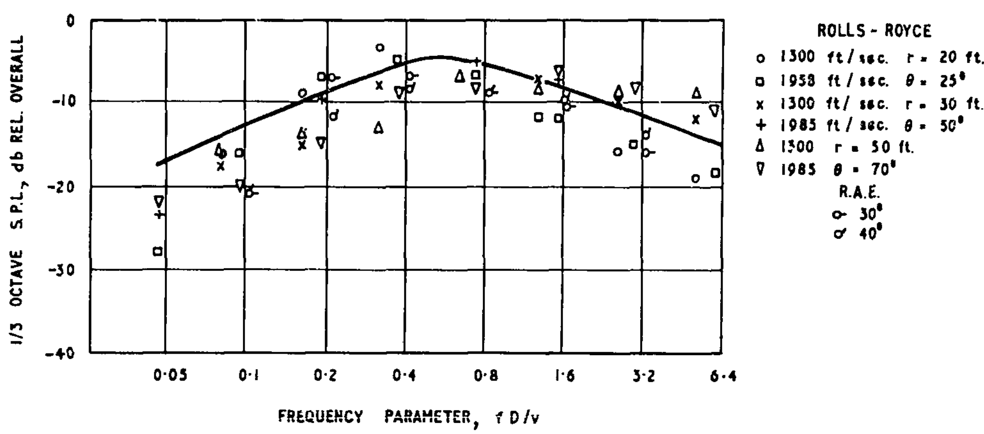


Fig.17(b) Comparison of spectra 10 diameters downstream

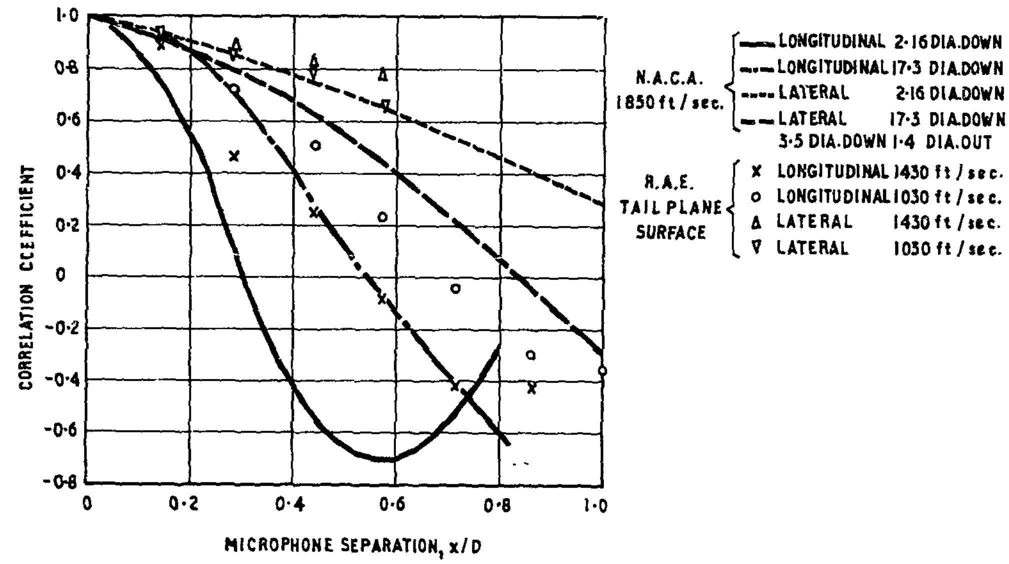


Fig.18 Comparison of space correlograms

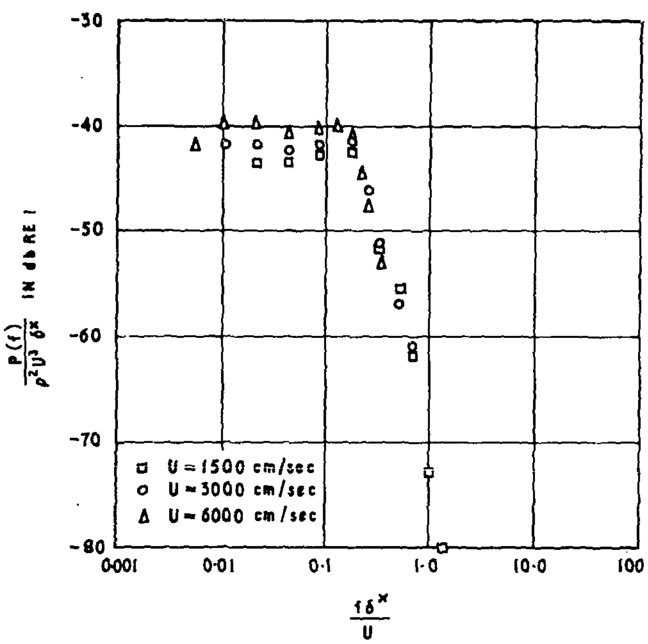


Fig.19 Boundary-layer wall pressure spectra (Ref.35)

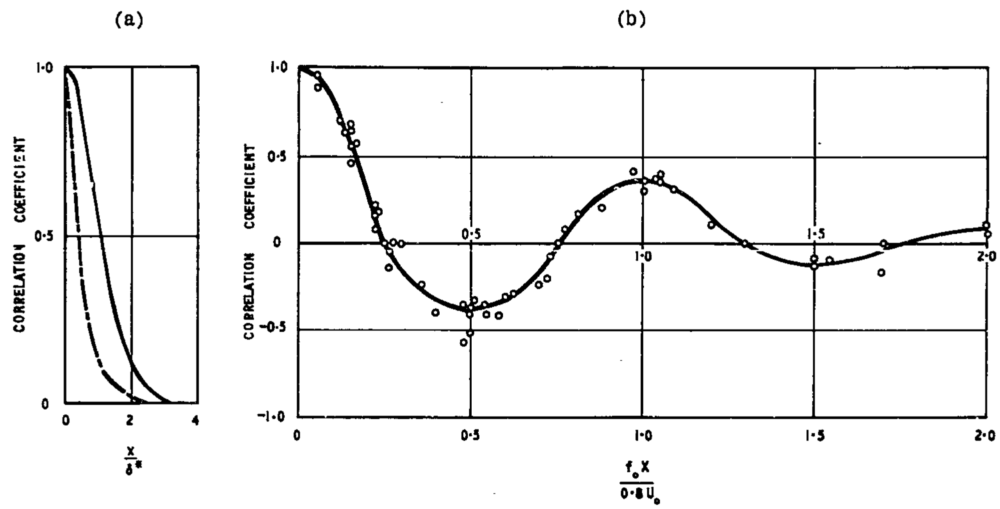


Fig.20(a) Overall longitudinal space correlogram of wall pressures (Ref.33)

(b) Filtered longitudinal space correlogram of wall pressures (Ref.35)

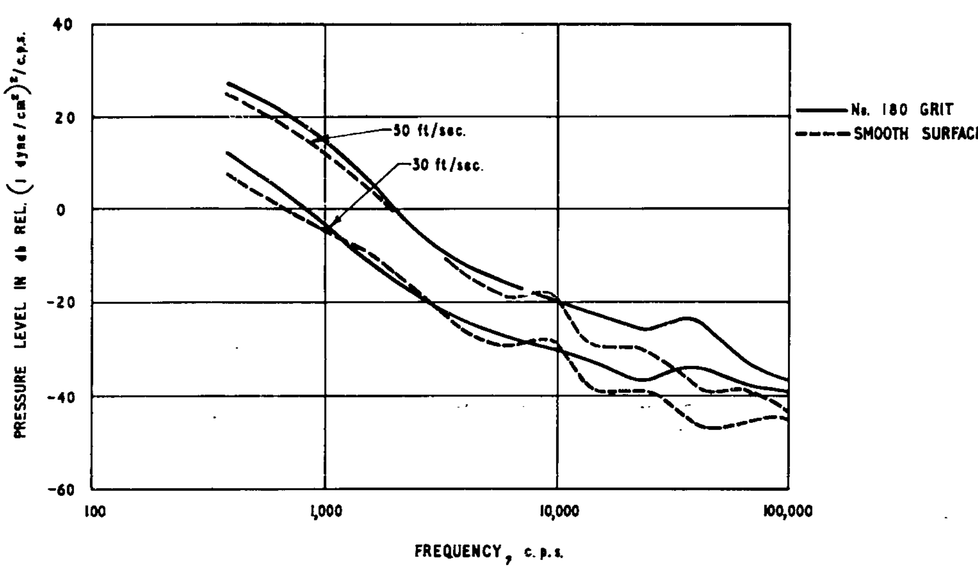


Fig.21 Effect of roughness on wall pressure spectra (Ref.42)

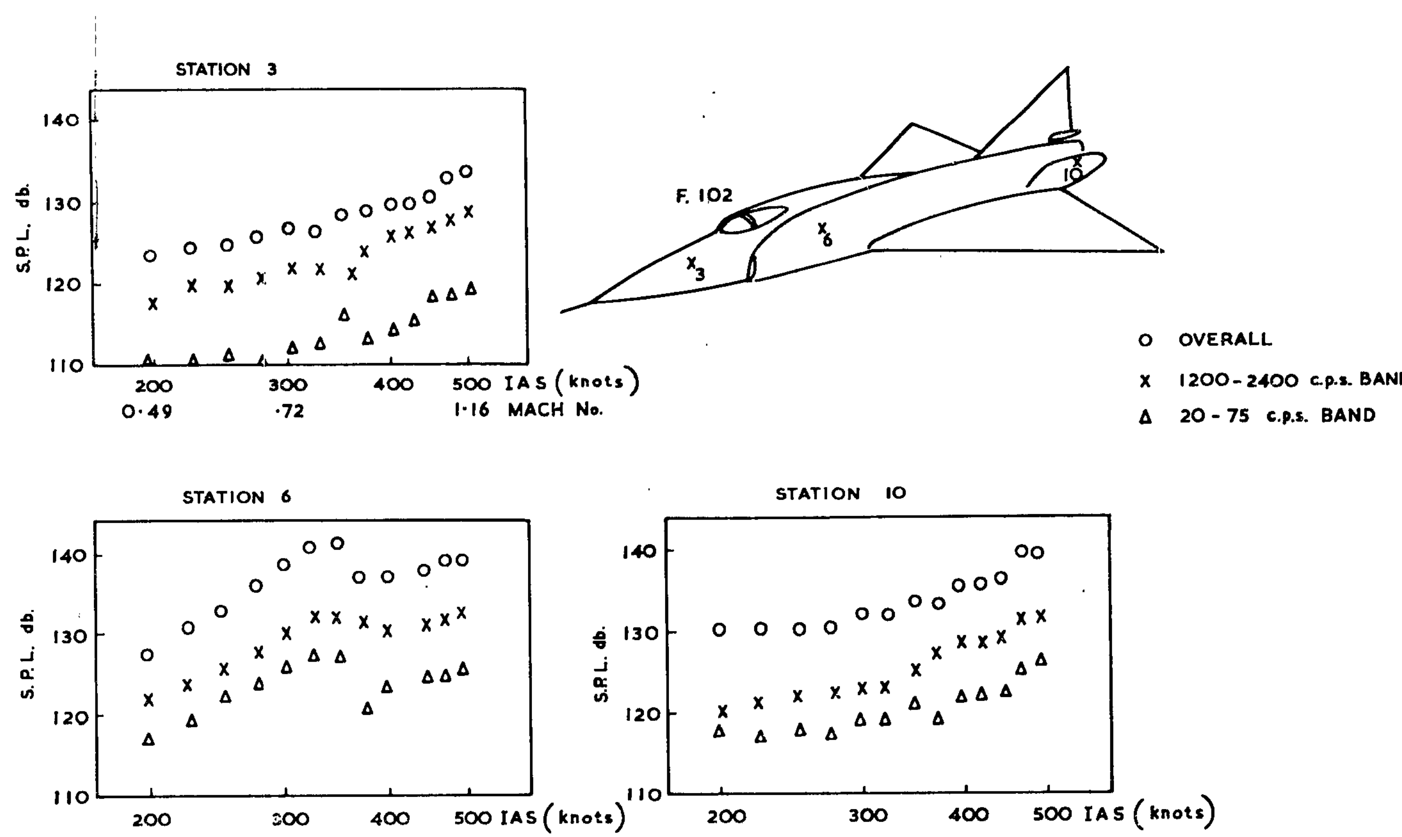


Fig.22 Variation of boundary-layer wall pressure with flight speed (Ref.19)

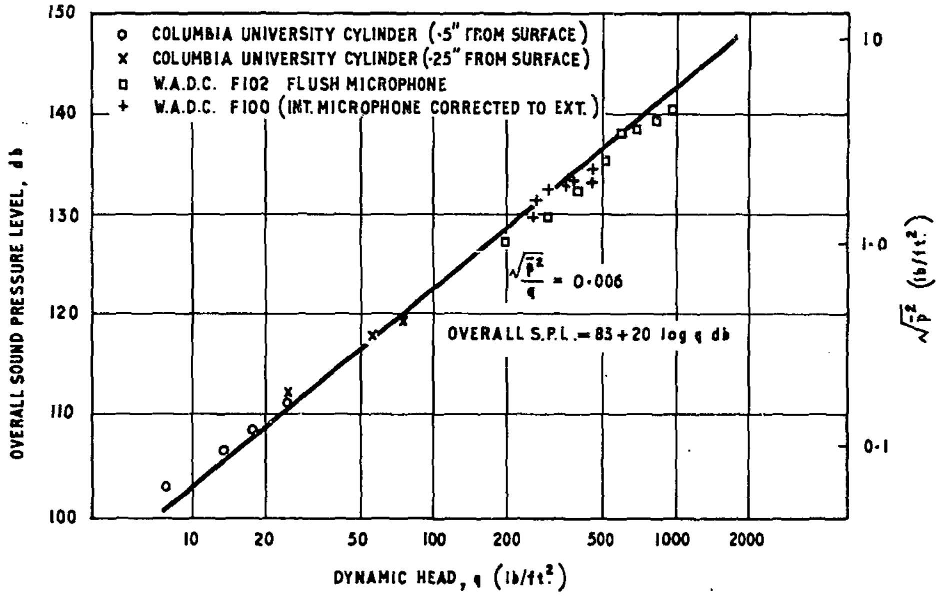


Fig.23(a) Variation of overall sound pressure level with dynamic pressure (Ref.19)

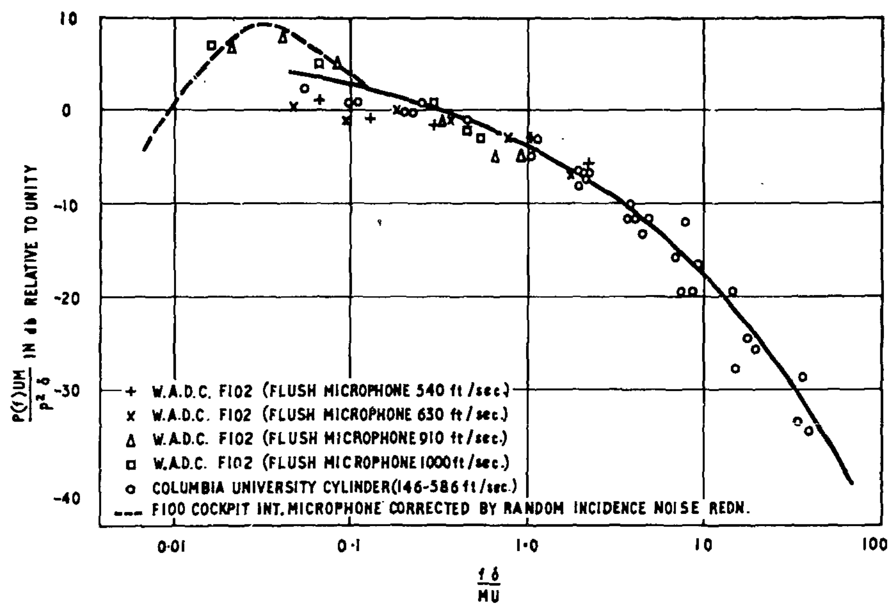


Fig.23(b) Spectra of boundary-layer pressure fluctuations (Ref.19)

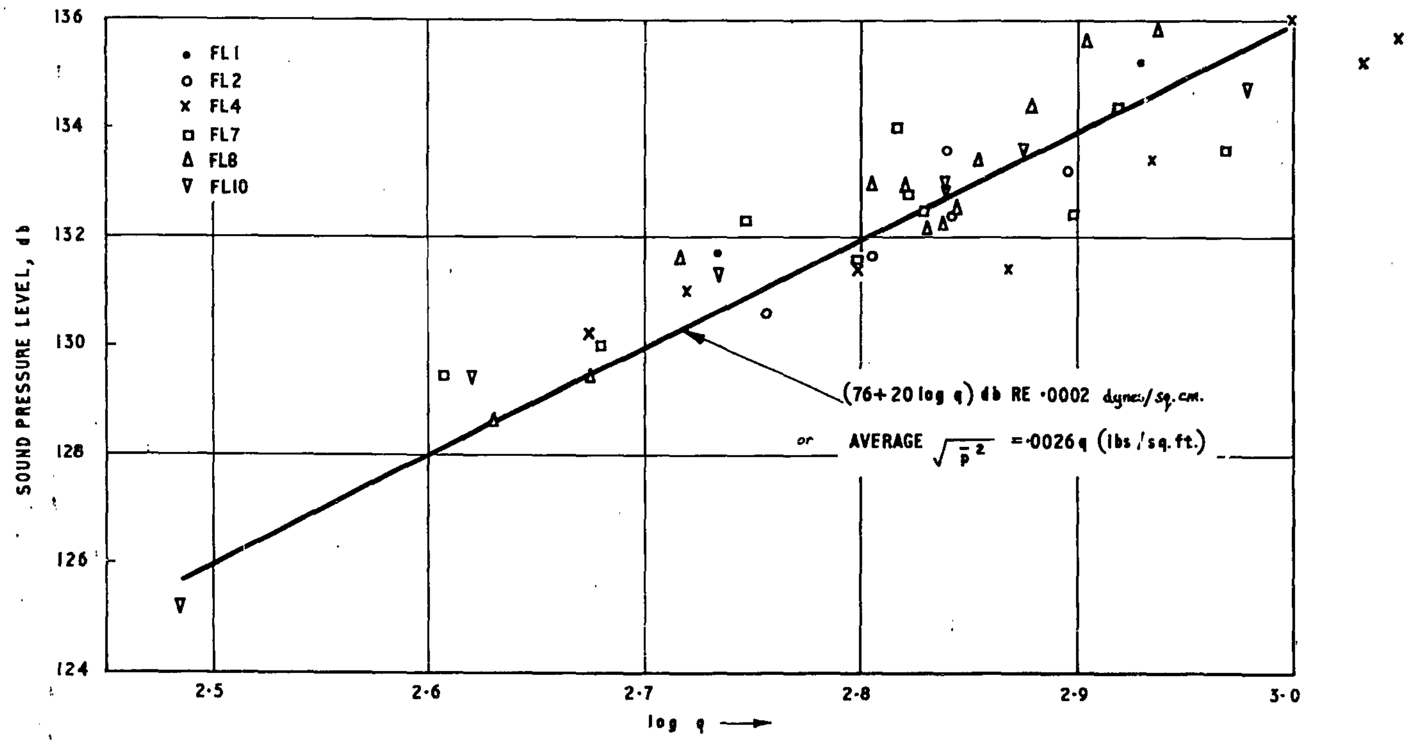


Fig.24 Variation of overall pressure levels on under wing surface with dynamic pressure (Ref.51)

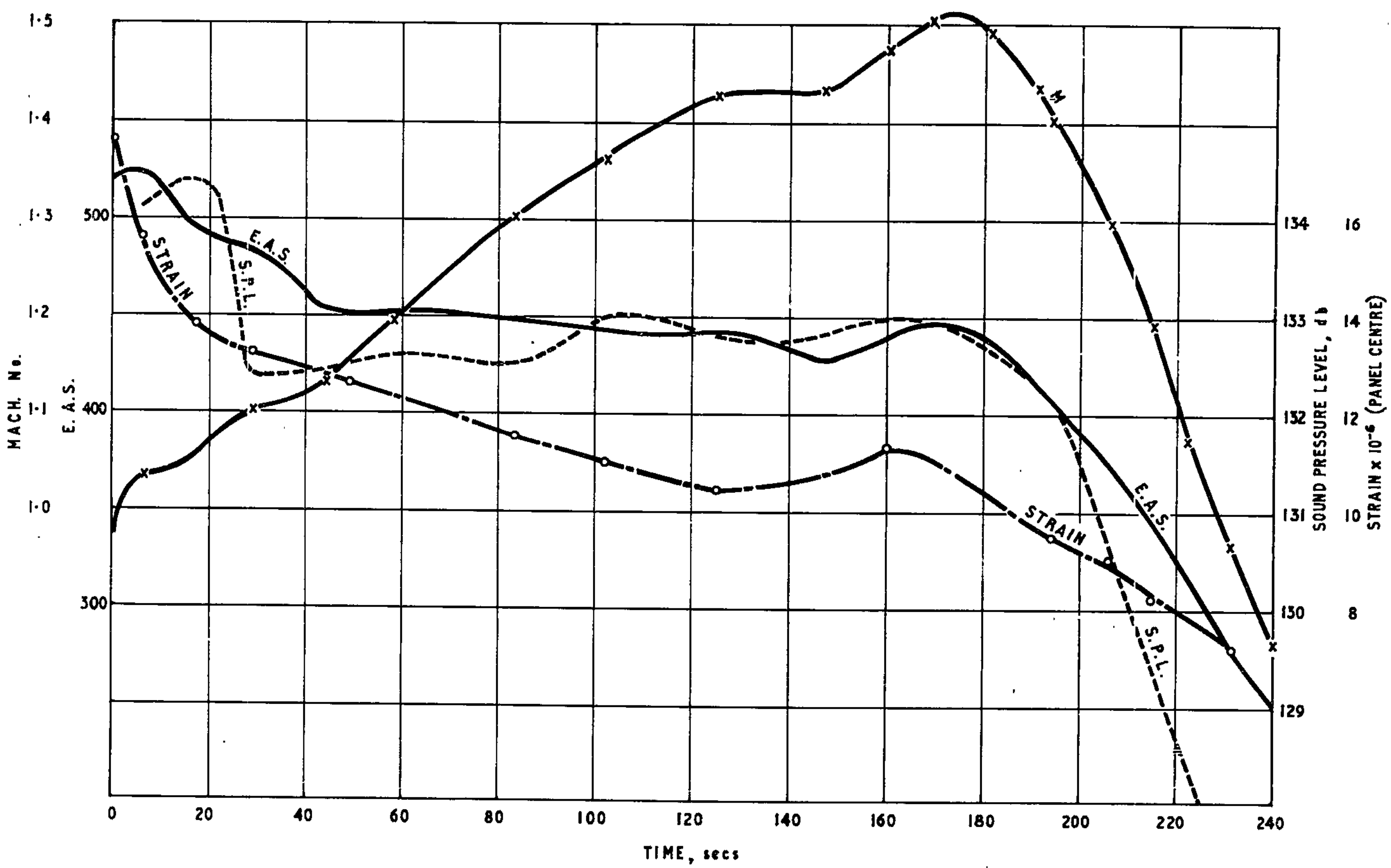


Fig.25 Sound pressure level and panel strain for flight in constant E.A.S. (Ref.51)

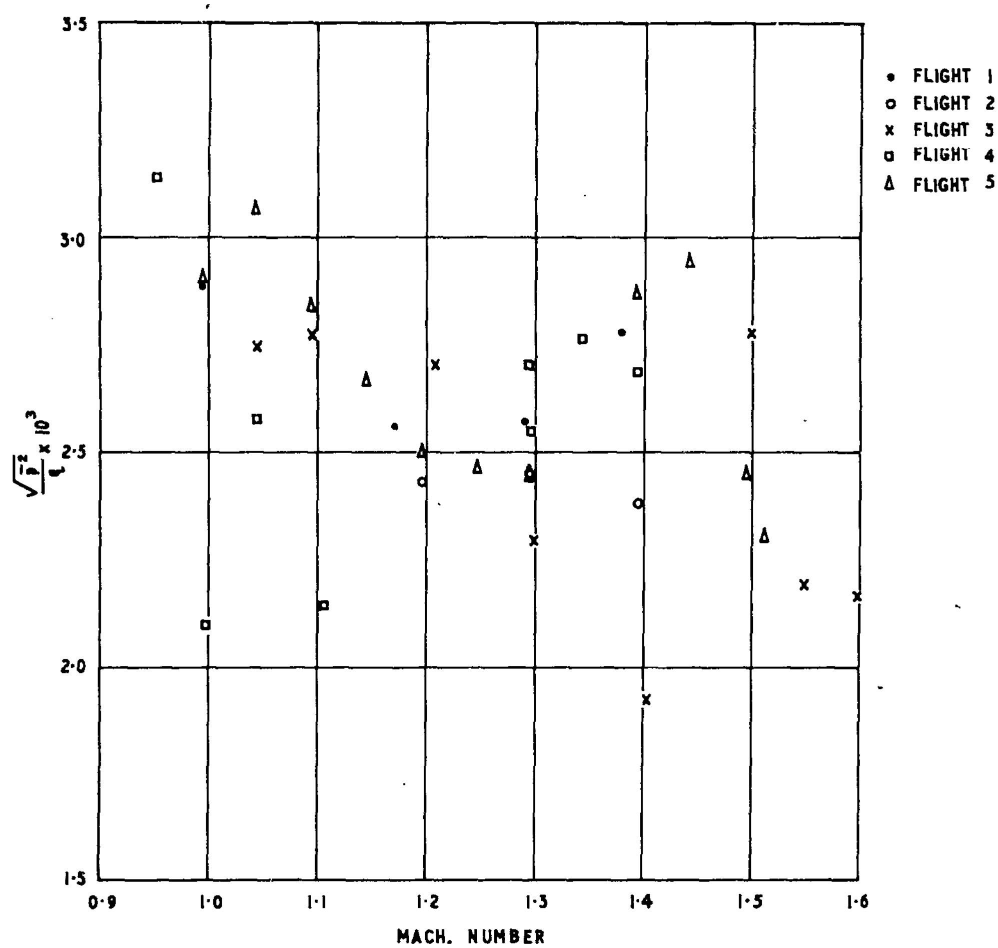


Fig.26 Variation of sound pressure level with Mach number (Ref.51)

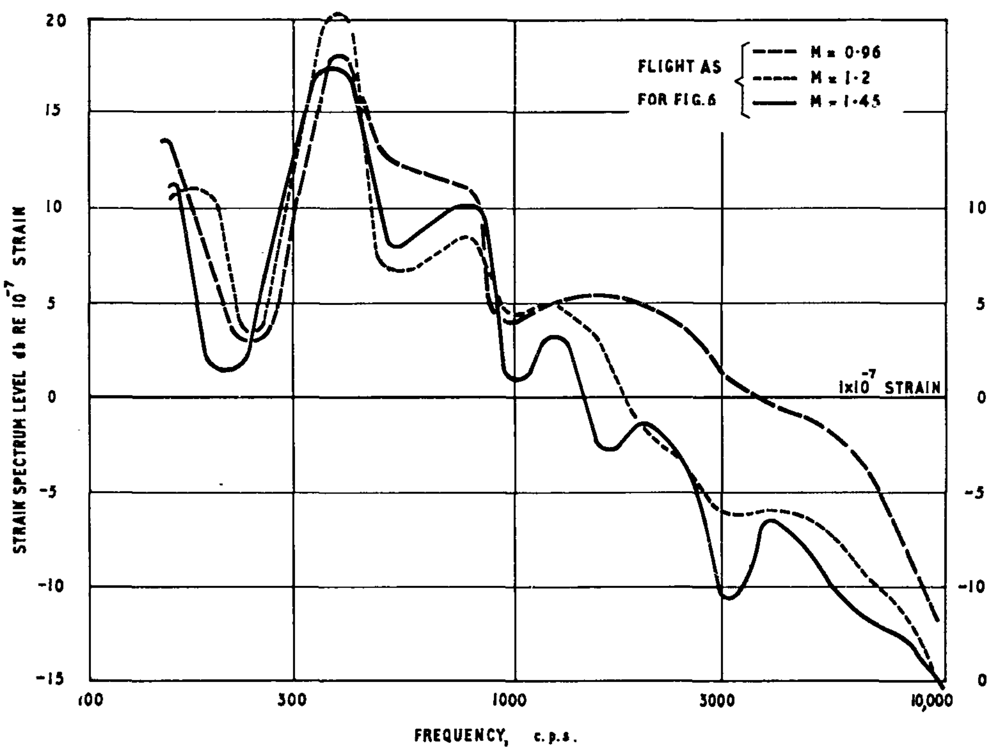


Fig.27 Strain spectra - boundary-layer excitation (Ref.51)

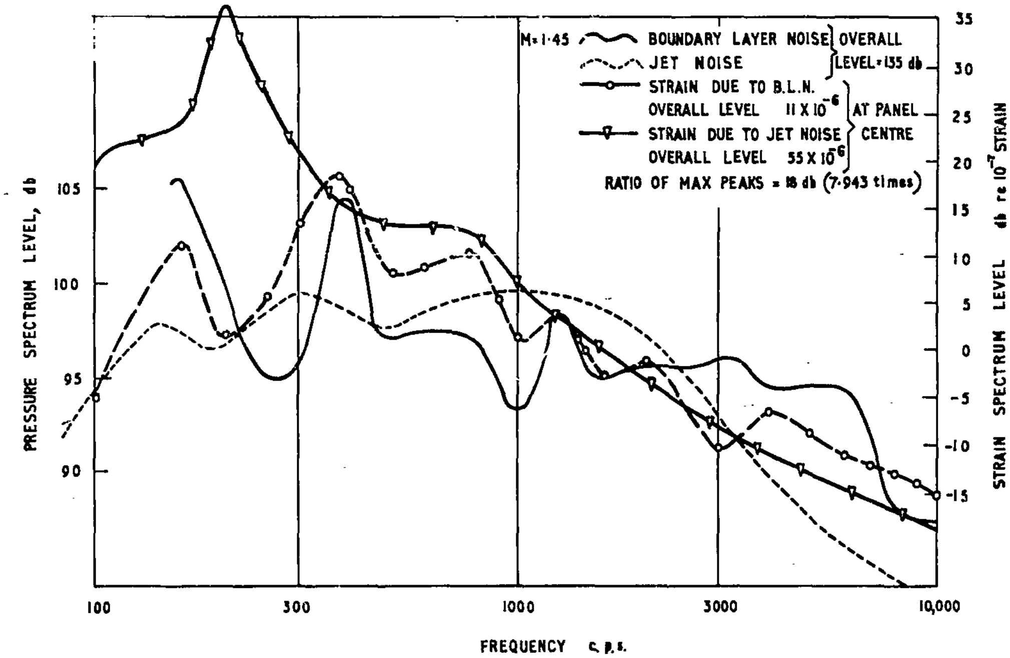


Fig.28 Comparison of strain spectra for jet and boundary-layer excitation (Ref.51)

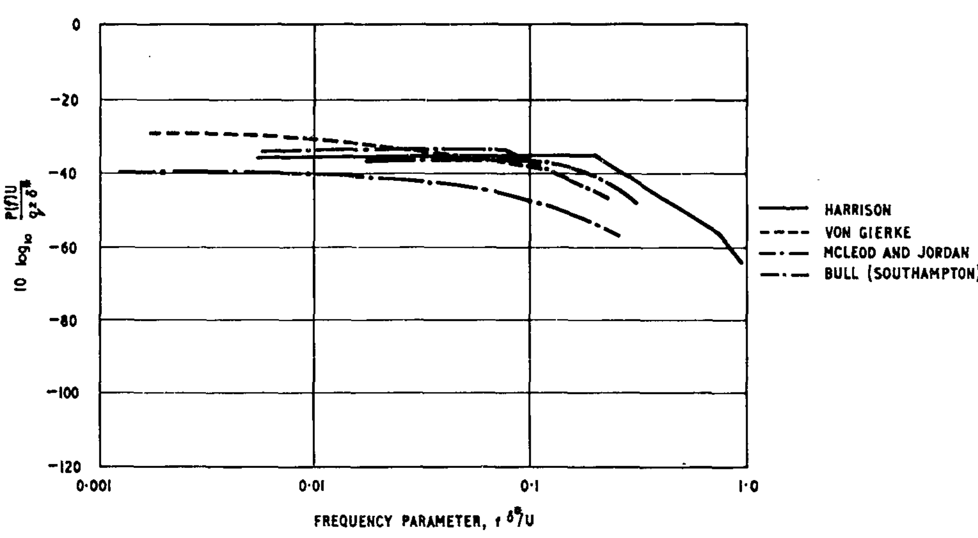


Fig.29 Comparison of measured spectra

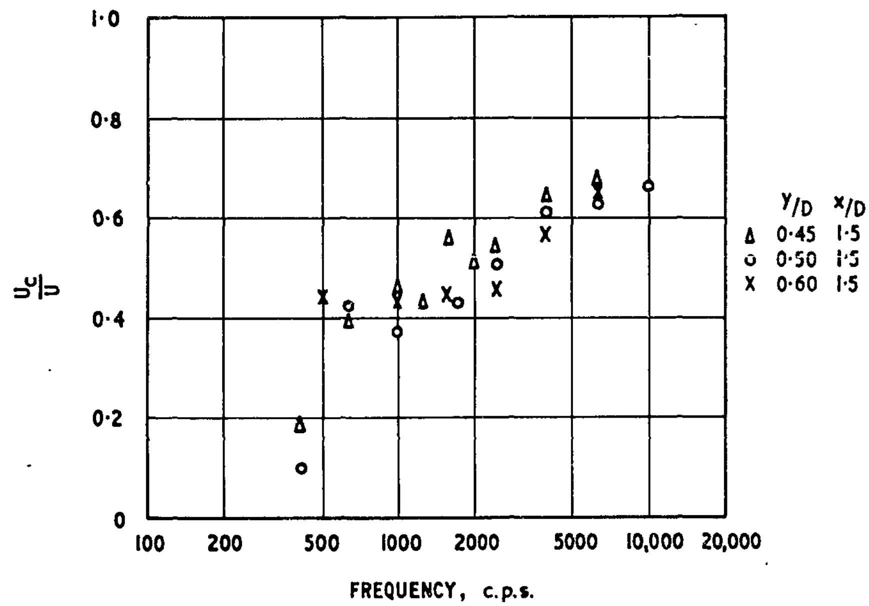


Fig.30 Convection velocities of turbulence in a 1 inch diameter jet

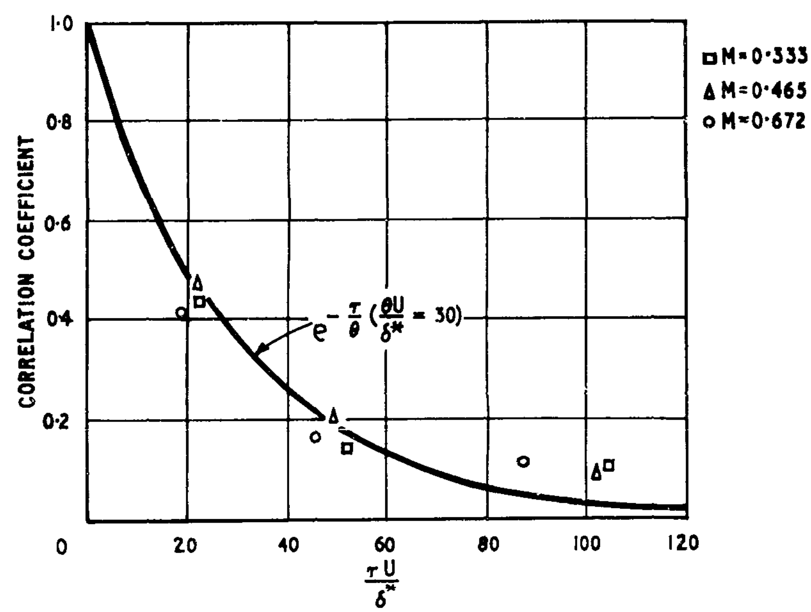


Fig.31 Autocorrelation in moving axes at subsonic speeds (Ref.34)

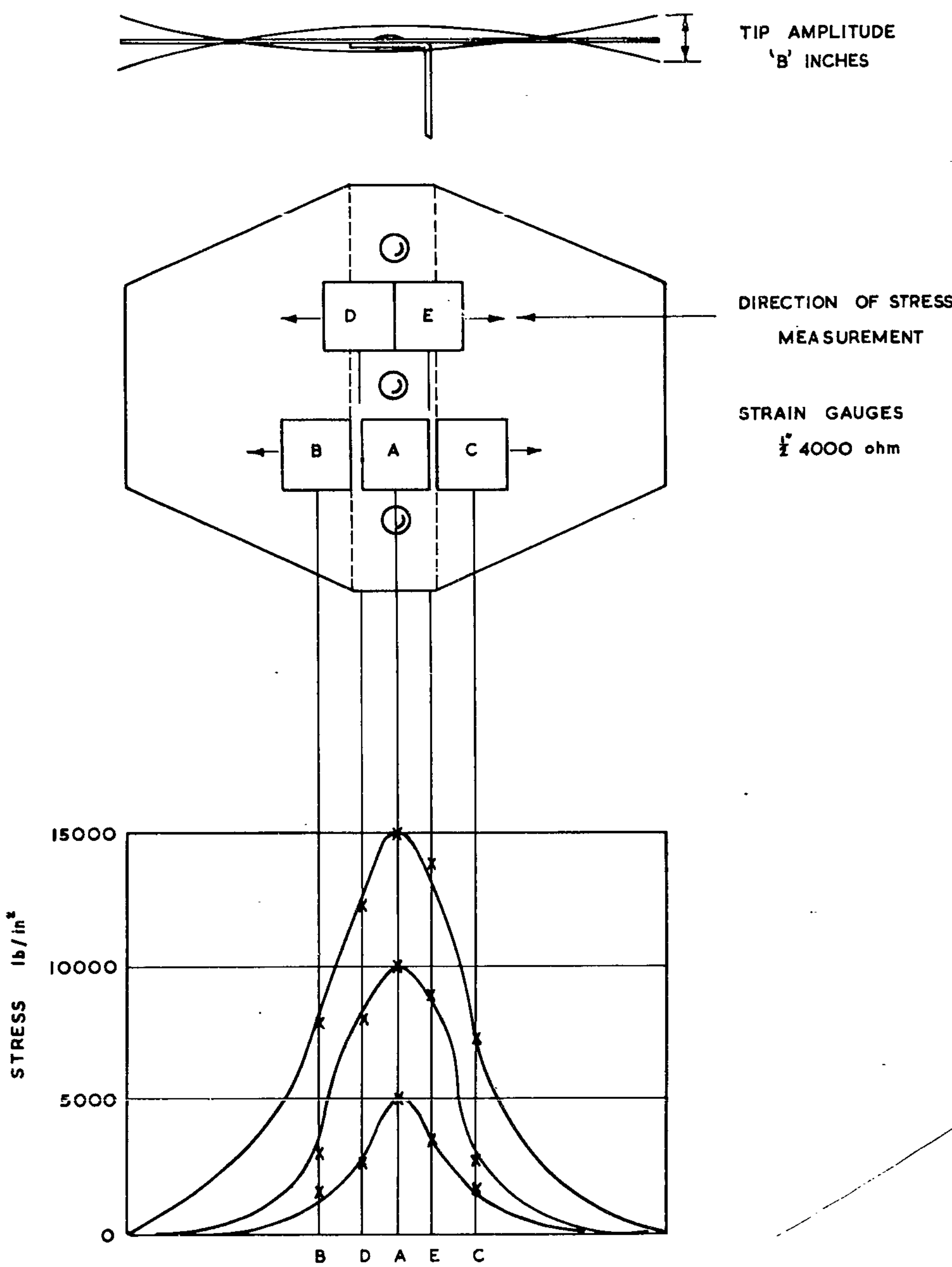


Fig.32 Strain gauge positions and stress distribution (Ref.59)

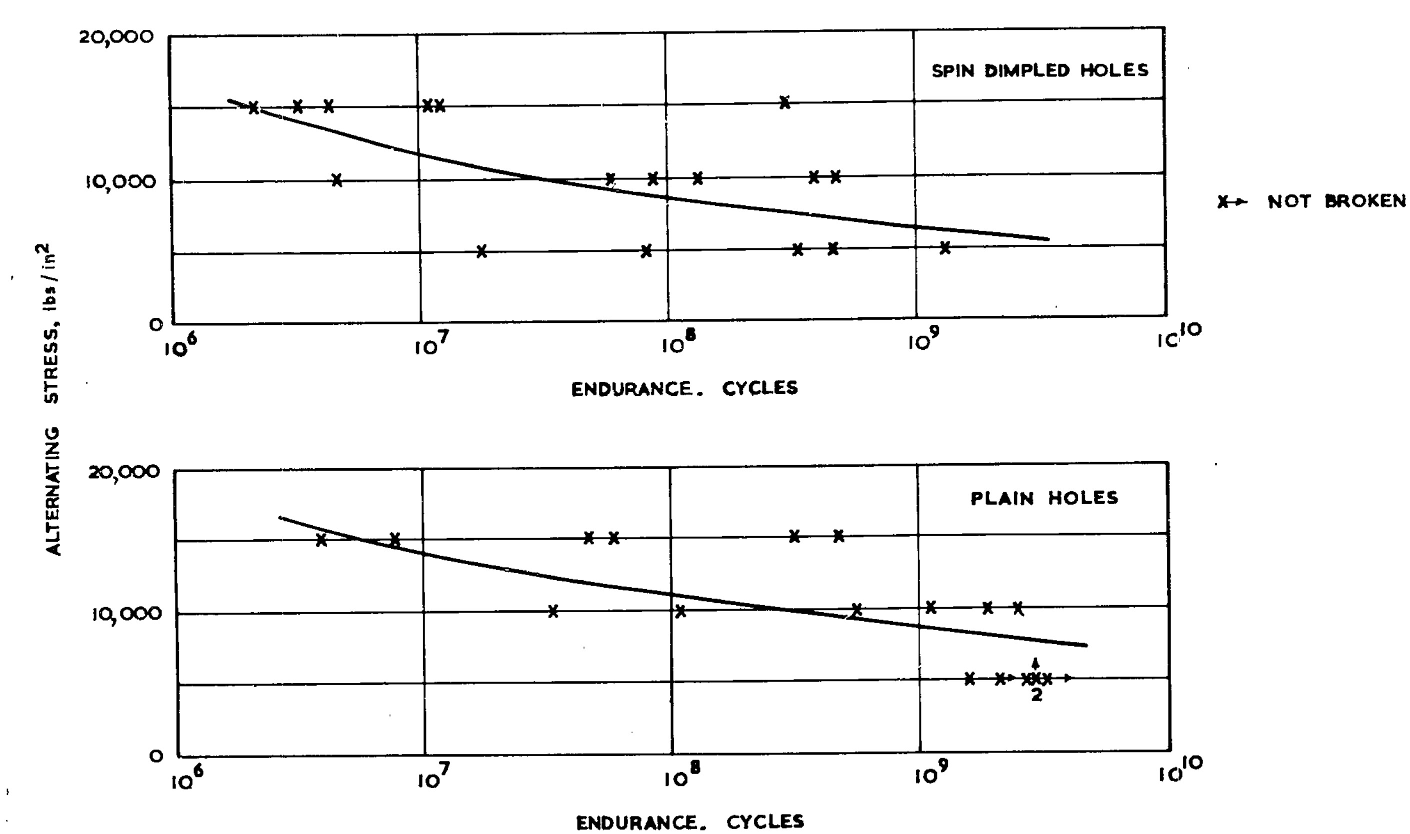


Fig.33 S-N curves, DTD 710 material (Ref.59)

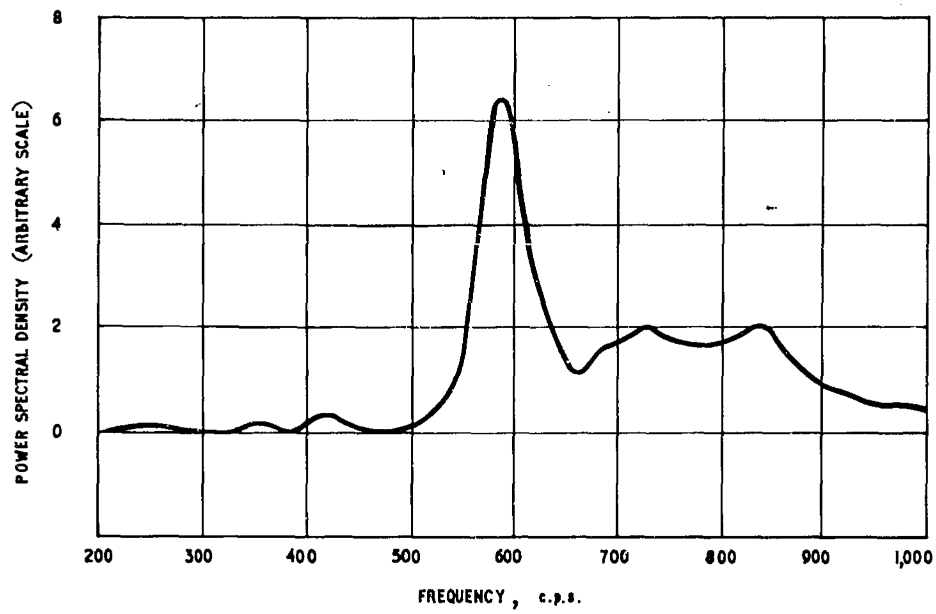


Fig.34 Strain spectrum of Caravelle fuselage skin panel (Ref.62)

FIGURE 35

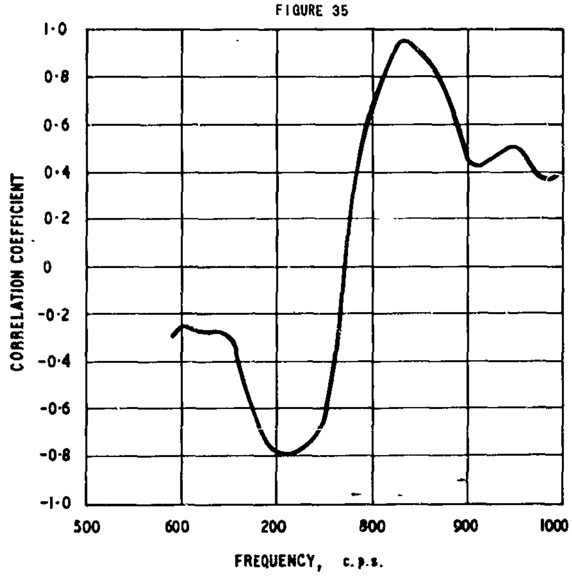


FIGURE 36

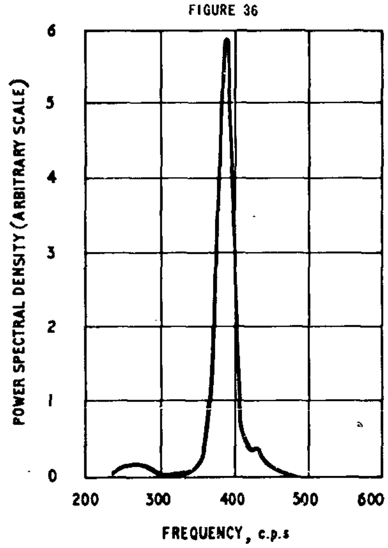


Fig.35 Correlation spectrum of adjacent panel strains (Ref.62)

Fig.36 Strain spectrum of Comet tailplane skin panel (Ref.64)

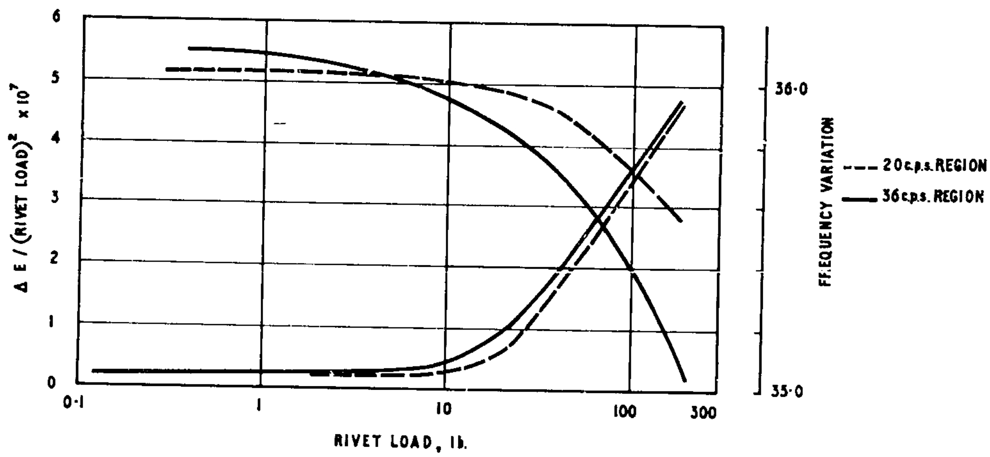


Fig.37 Variation of energy dissipation per cycle at a riveted joint with load amplitude (Ref.65)

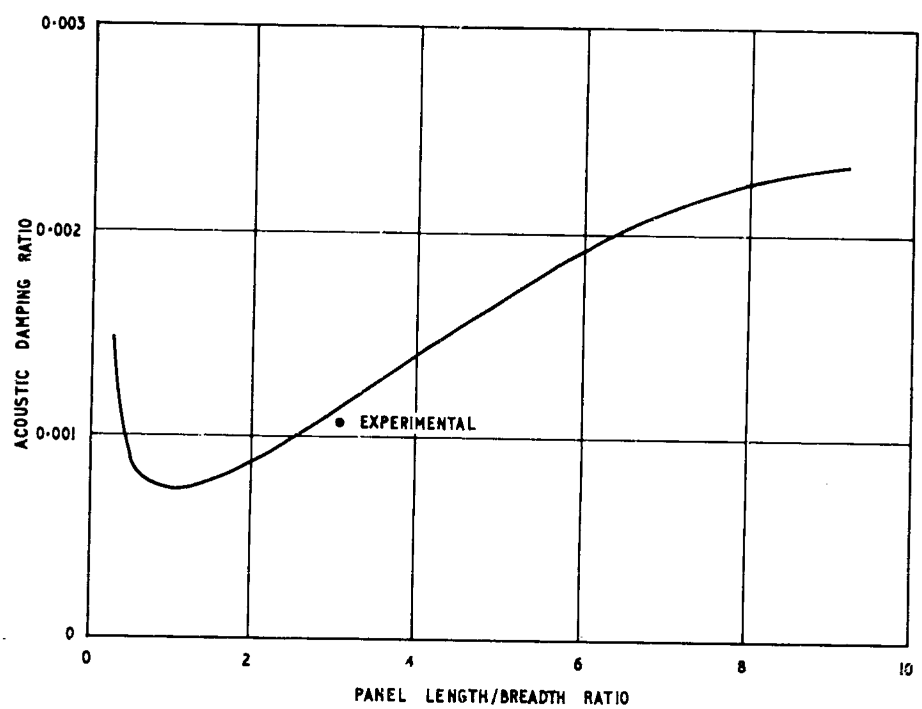


Fig.38 Acoustic damping of a simply-supported flat steel panel 0.048 in. thick in a rigid baffle

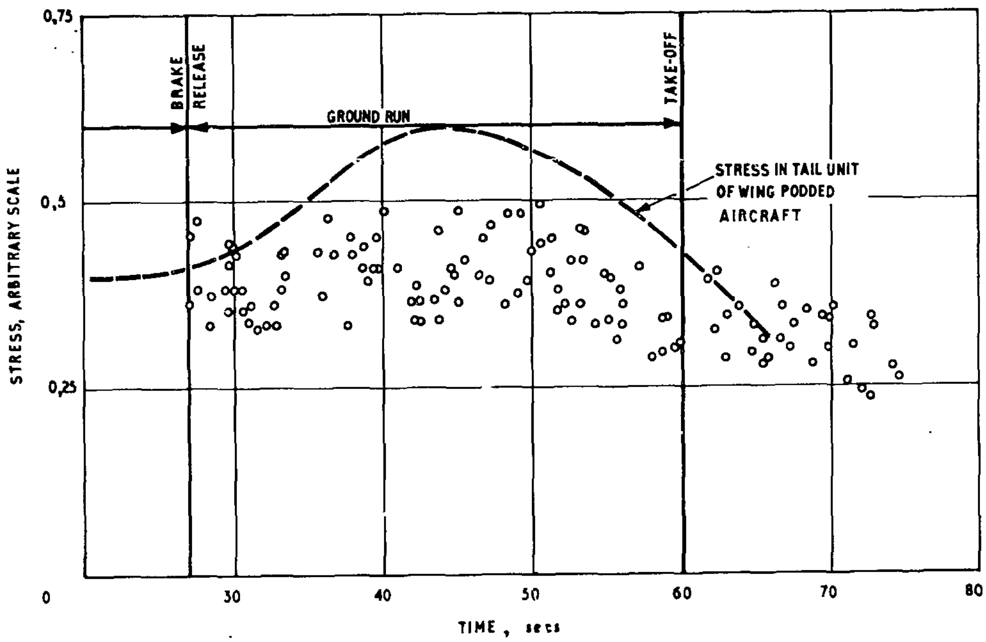


Fig.39 Variation in strain during ground run - Caravelle (Ref.69)

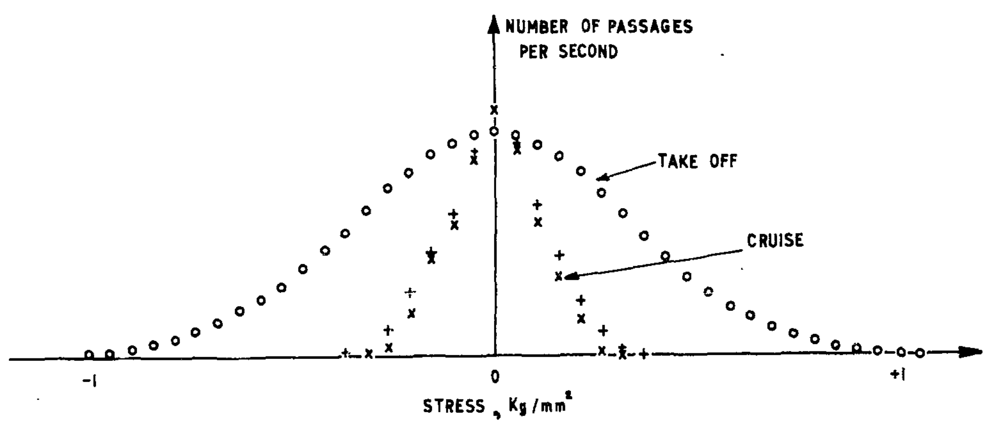


Fig.40 Distribution of stress amplitude at take-off and cruise - Caravelle (Ref.69)

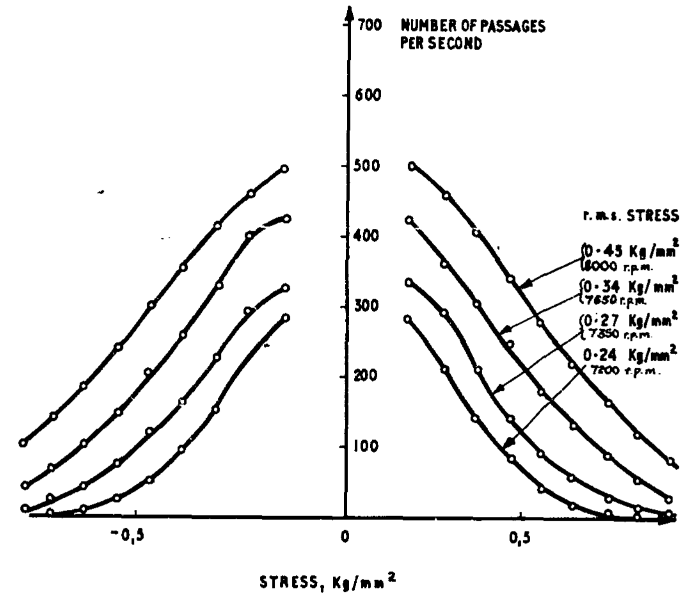


Fig.41(a) Distribution of stress amplitude for different engine speeds - Caravelle (Ref.69)

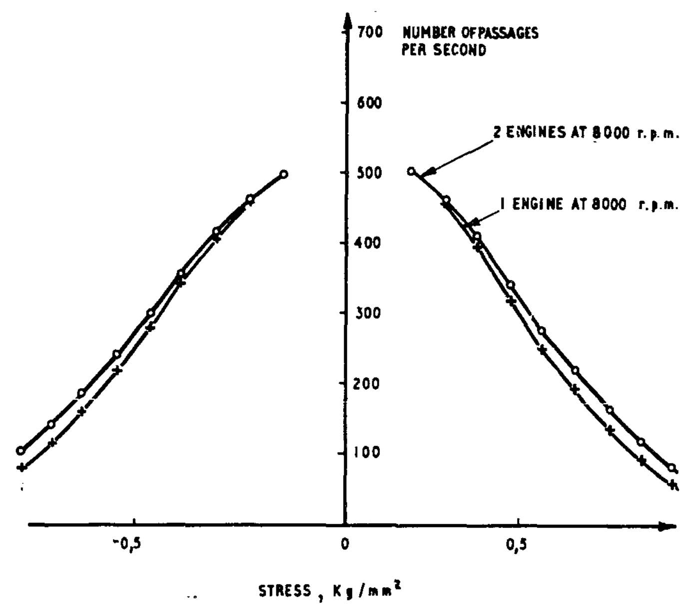


Fig.41(b) Distribution of stress amplitude for 1 engine and 2 engine operation - Caravelle (Ref.69)

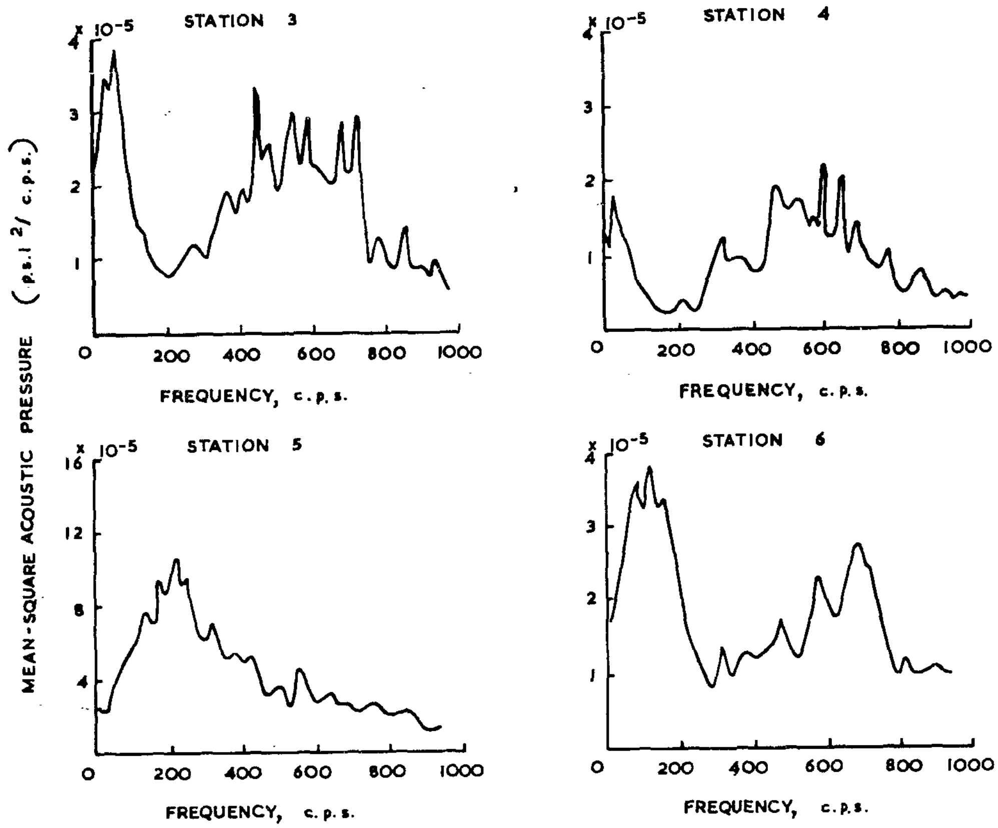


Fig.42 Power spectra of noise pressures on panels (air jet) (Ref.72)

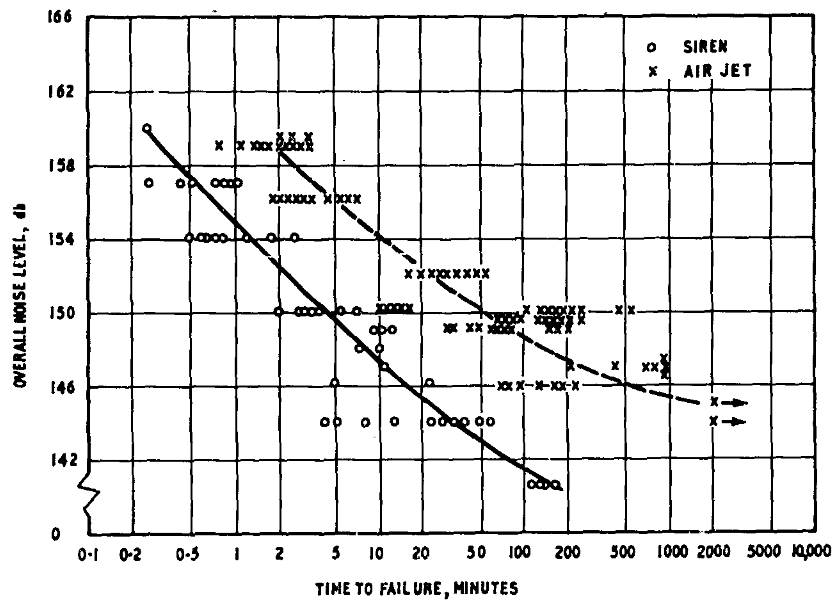


Fig.43(a) Fatigue life of 0.032 in. panels as a function of overall noise level for random and discrete loading (Ref.72)

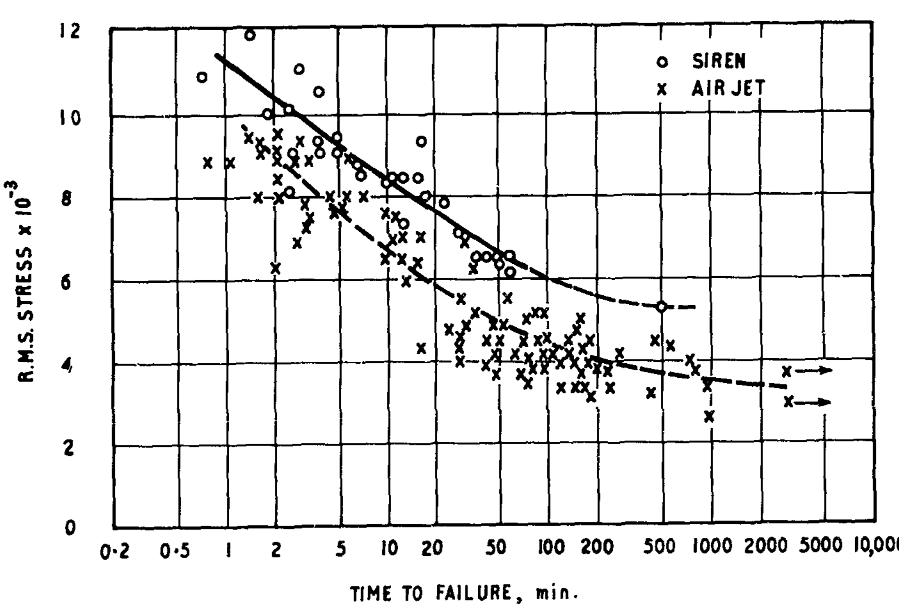


Fig.43(b) Fatigue life of 0.032 in. panels as a function of r.m.s. stress for random and discrete loading (Ref.72)

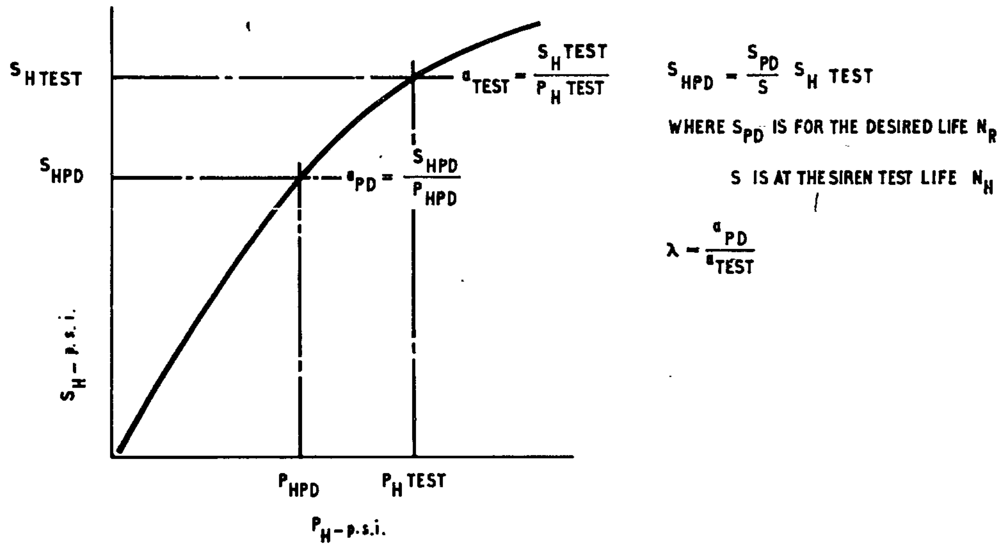


Fig.44(a) Computation for non-linearity factor (Ref.85)

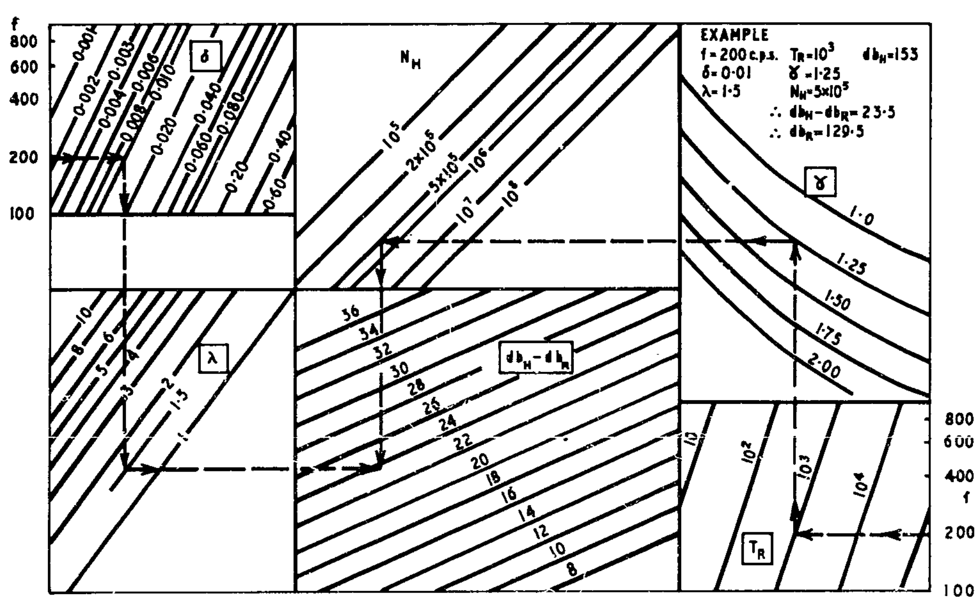


Fig.44(b) Conversion chart: discrete to random loading (Ref.85)

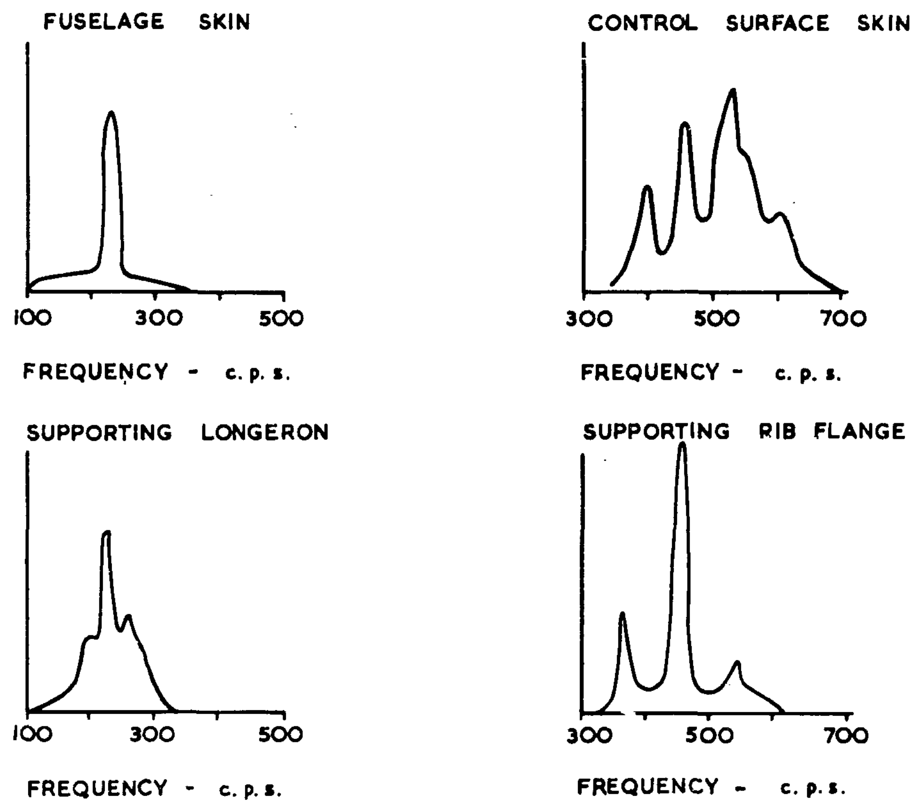


Fig.45 Measured power spectra of strains in different parts of RB-66 structure (Ref.31)

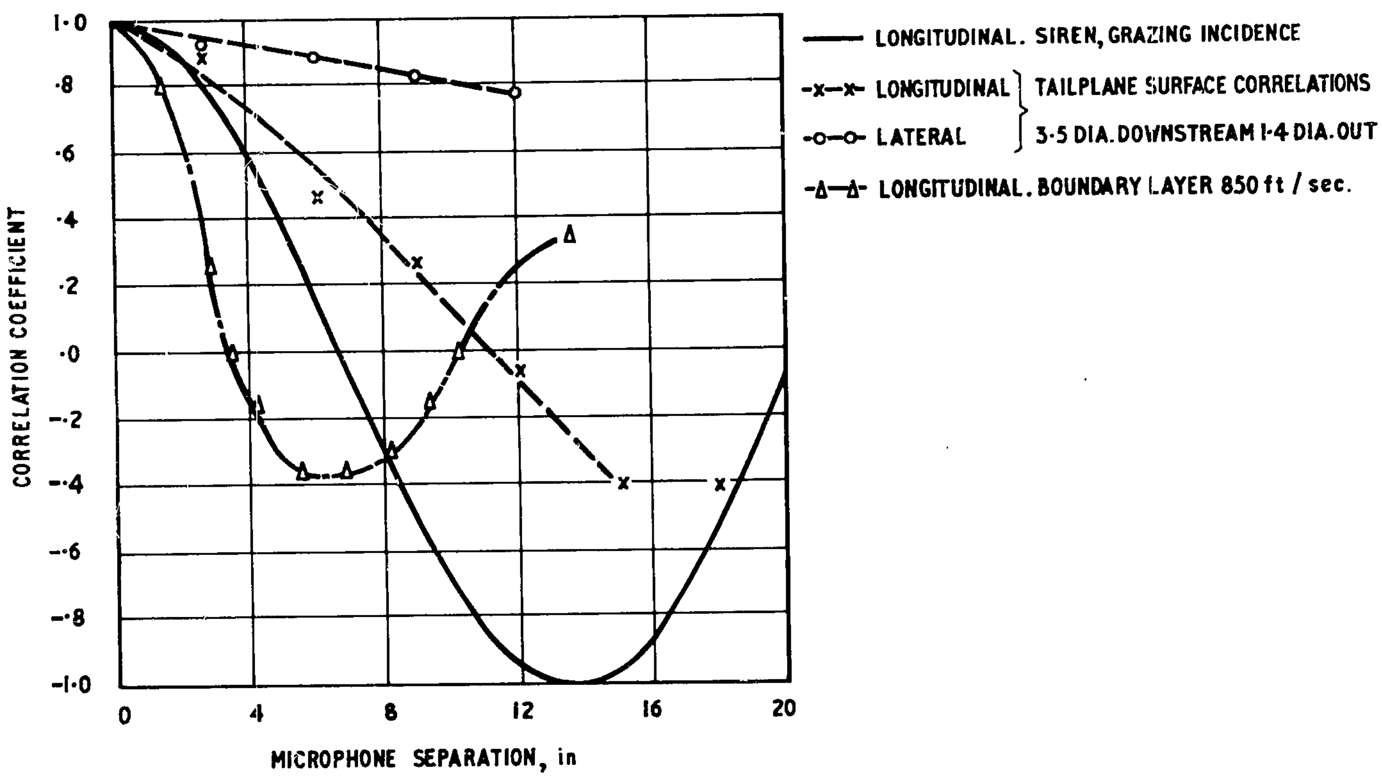


Fig. 46 Comparison of jet, boundary layer, and siren filtered space correlograms at 500 c.p.s.

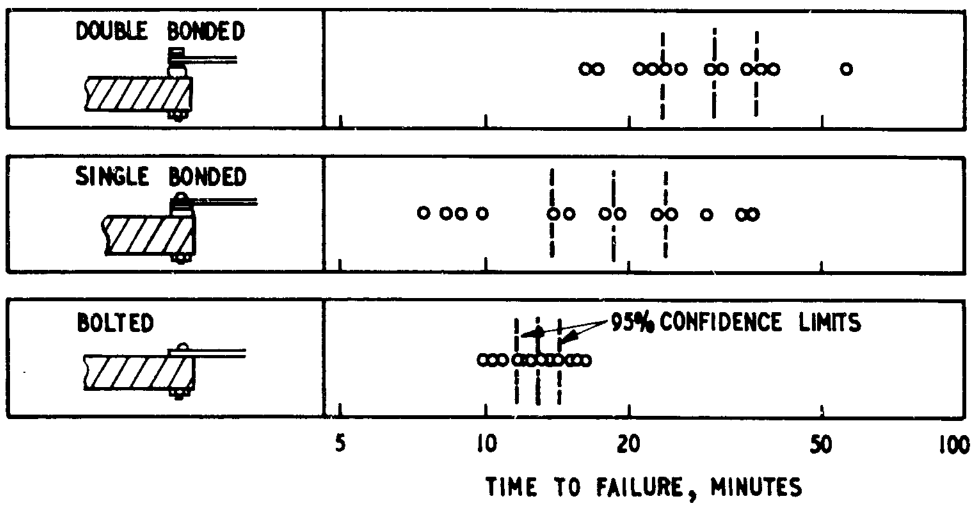


Fig.47 Effect of edge fixing in panel life (Ref.72)

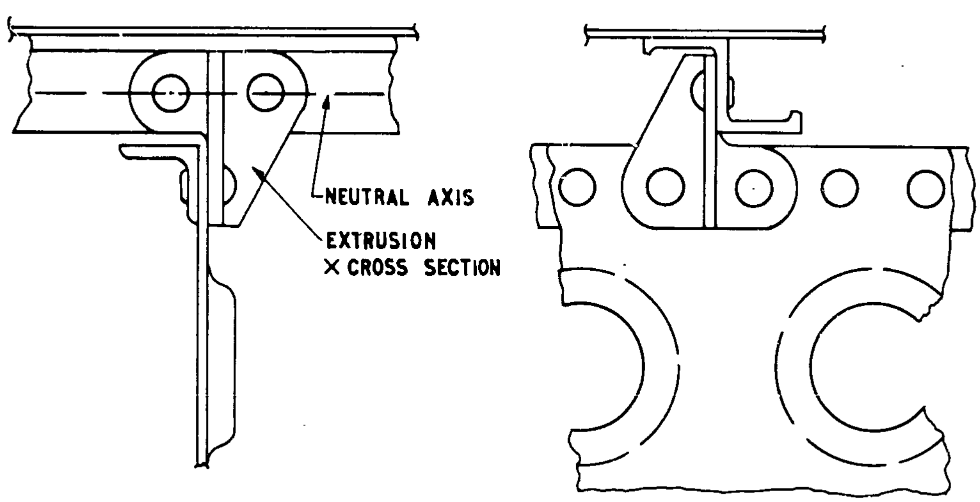


Fig.48 Detail DC-8 design of stringer to rib attachment (Ref.31)

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Section II deals with boundary-layer pressure fluctuations and suggests that the available information establishes that up to transonic speeds the skin root-mean-square pressure level is approximately equal to 0.006q. It is thought that in all probability the spectrum is sufficiently well defined for structural response estimates.

In Section III the studies involving the evaluation of response to random pressure fluctuations are considered and it is concluded that much more experimental work will be necessary before the various theories developed can be put to practical application.

A list of specific areas recommended for further research is included.

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